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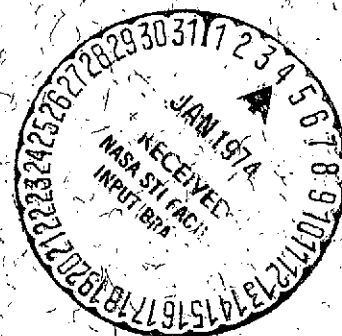
HISTORY OF SAN MARCO

(NASA-TM-X-70523) HISTORY OF SAN MARCO
(NASA) 76 p HC \$6.00
72 CSCL 04A

N74-13091

G3/13 Unclass
24500

DECEMBER 1968



GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

HISTORY OF SAN MARCO

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Explorer and International Project
Management Offices

December 1968

GODDARD SPACE FLIGHT CENTER
Greenbelt, Maryland

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SUMMARY

This document is a brief history of the first San Marco project, a joint program of the United States and Italy for investigating upper-air density. The San Marco project is part of the effort of the National Aeronautics and Space Administration (NASA) Office of International Programs to cooperate with other nations in peaceful aspects of space exploration. Cooperative efforts of this type provide NASA with the opportunity to work with scientists of other nations, not only in sharing the knowledge gleaned from our space efforts and assisting them to organize and pursue their own special lines of study, but also in using the knowledge of their scientists, the skill of their engineers and technicians, and the resourcefulness of their industry in meeting the challenges of space.

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HISTORY OF SAN MARCO

1. INTRODUCTION

1.1 PURPOSE

The purpose of this document is to provide a brief history of the initial San Marco project from 1962 to 1967, and to record the project support provided by the Goddard Space Flight Center (GSFC).

1.2 SUMMARY

The San Marco project¹ was a three-phase cooperative effort of the United States and Italy to investigate upper-air density and associated ionosphere phenomena. The initial phase included the design and development of the spacecraft, the experiments, the launch complex, and a series of suborbital flights from Wallops Island (WI). The suitability of the atmospheric-density experiment and spacecraft instrumentation was confirmed by two suborbital launches from WI, using Shotput vehicles. Also, during the initial phase, a series of Nike-Apache rockets was launched from the Kenya complex for validation purposes.

The second phase, consisting of designing, fabricating, and testing a spacecraft for the first orbital mission, culminated in an orbital launch also from WI. The launching of San Marco A by an Italian crew, using a Scout vehicle, proved out the overall system concept and provided the crew with the training and experience required for the third-phase equatorial launch.

The third phase consisted of further refining the experiments and spacecraft instrumentation and of establishing a full-bore Scout complex in Kenya.

The launch of San Marco B, in April 1967, from this complex into an equatorial orbit, concluded the initial San Marco effort.

During the subsequent year, a new agreement was made to continue the investigation of the atmosphere by adding two Goddard experiments on-board the San Marco C spacecraft.

1.3 RESPONSIBILITIES

Responsibilities for the program appear in an official agreement between Italian Foreign Minister Pietro Piccioni and Vice President Lyndon B. Johnson,

dated September 5, 1962, and in a memorandum of understanding signed May 31, 1962, by Professor Luigi Broglio, Director of the Aerospace Research Center (CRA) of the University of Rome, and Dr. Hugh L. Dryden, Deputy Administrator of NASA.

The agreements (Appendixes A and B) define the goals and restraints of the program and the commitments of the cooperating agencies. In general, they provided for a three-phase program in which CRA was responsible for:

- Designing, fabricating, and testing all payloads
- Providing a suitable equatorial launch complex
- Providing tracking, data-acquisition, and data-reduction facilities not available from NASA
- Supporting CRA personnel during training
- Performing necessary studies to ensure acceptable transport and handling of the third-phase Scout vehicle

NASA was responsible for:

- Providing launch vehicles (sounding rockets for the first phase and Scout boosters for the second and third phases)
- Training spacecraft engineers and supplying technical consultation (including data to facilitate effective payload design, fabrication, and testing)
- Providing additional spacecraft training as required
- Providing data-acquisition and tracking services
- Training and consultation in data acquisition and reduction
- Training personnel for vehicle assembly, launch, and range safety

2. SAN MARCO PROJECT

2.1 OBJECTIVES

While the first San Marco program was being proposed, several objectives became evident. Although they were part of an overall concept, these objectives were diversified with respect to time and essence; i.e., one was short-ranged and scientific; the other, long-ranged with operational goals in future space work.

2.1.1 Scientific

The scientific objective of the project was to obtain information on the atmosphere in the equatorial region, 120 to 200 miles high. Because this region is above the useful range of sounding rockets and below a practical operating limit for long-lived satellites, few data exist on air density and molecular temperature. An equatorial orbit was selected because:

- The greatest solar-terrestrial effects are believed to occur in this region.
- The latitude-variation effect is eliminated.
- The low-inclination orbit, coupled with a shifting perigee and a short period (≈ 90 minutes), permits the acquisition of much data over the same locations (≈ 15 orbits per day).

Some examples of the phenomena caused by solar activity are:

- Diurnal variation (diurnal bulge), caused by the earth's rotation, occurs in such a manner that maximum density exists at a point about 2 hours behind the suborbital point. Therefore, the surfaces of equal density possess a characteristic pear shape.
- Monthly variation resulting from the sun's rotation about its axis every 27 days brings solar-activity centers into view.
- Semiannual variation caused by the solar wind and the position of the geomagnetic field relative to the sun
- Random short-period variations caused by solar flares and the ensuing geomagnetic storms

The method now used for observing cumulative changes in a satellite's orbital period over many orbits suffices for measuring the monthly and semiannual variations. This technique, however, cannot be used for short-period fluctuations. The San Marco approach affords an immediate indication of solar-flare-induced changes and should give a detailed picture of the diurnal bulge.

Two approaches were used for investigating ionosphere characteristics:

- Long-range propagation observations
- Electron-content measurements

Long-range propagation with a satellite beyond a receiving station horizon is sometimes observed. Several propagation mechanisms may be responsible for this phenomenon: Apart from hop propagation, the most interesting phenomenon is guided propagation in which a radio wave travels inside the ionosphere ducted by suitable variations of the refraction index with height. The possibility of receiving a satellite signal "guided in an ionospheric duct" at a ground station is related to the existence of a gate caused by inhomogeneity in the ionosphere through which the signal can reach the ground. Information about ducting may lead to a practical use such as long-range radio transmission at low-power levels.

The electron content from the satellite to the ground was measured by the Faraday rotation method.

2.1.2 Operational

The operational objective of the project was to establish a launch capability in an equatorial region for launching sounding-rocket and Scout-size payloads. Essentially, the type of launch facilities at WI would have to be built and staffed by specially trained personnel.

2.2 EXPERIMENTS

2.2.1 Atmospheric-Drag Experiment

The primary experiment (Figure 1) carried by the San Marco satellites provided continuous direct measurement of the atmospheric drag and therefore of the atmospheric density in the altitudes from 200 to about 350 km. Before San Marco, the usual method of evaluating the atmospheric density was dependent on the determination of the so-called orbital acceleration, i.e., the time derivative of the orbital period. This quantity is exactly zero in an undisturbed

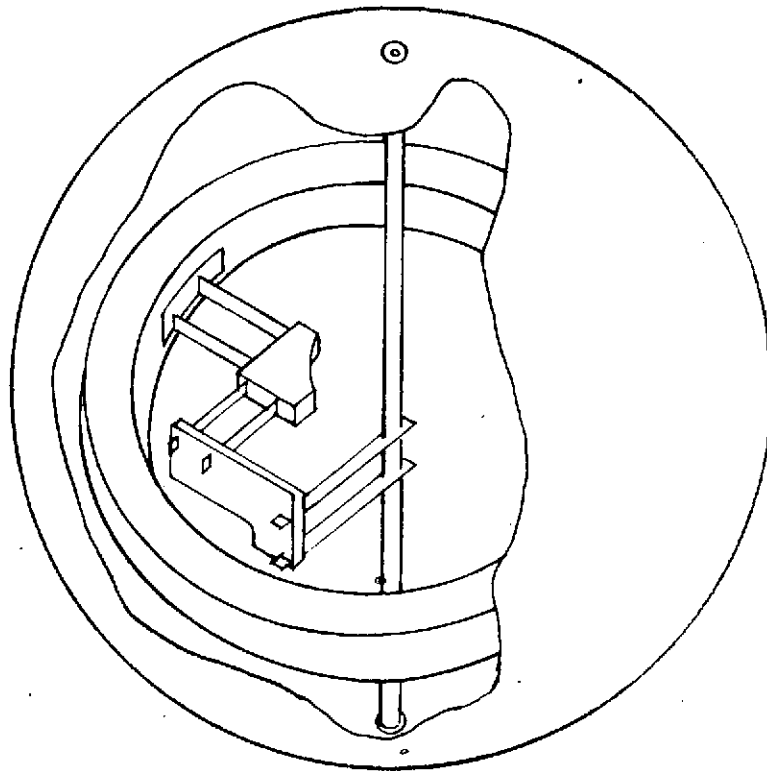


Figure 1. Atmospheric-Drag Experiment Schematic

orbital motion, but it assumes finite values in actual cases, depending on the atmospheric density. Because the period variations are extremely small, a large number of orbits must be tracked, and then, only an average value may be found.

This operation eliminates short-period variations such as daily density variations (atmospheric bulge) and those linked to the magnetic storms. However, the method of measuring air drag used by the San Marco satellite permitted a prolonged instantaneous density measurement. Based on a continuous measurement, this method also helps to determine the molecular temperature of the atmosphere and, with some assumptions, the mean molecular weight. In addition, the medium-inclination orbit for San Marco 2 permitted the measurement of the atmospheric density in the equatorial region where, until now, little data had been obtained.

To perform the atmospheric-drag measurement, the San Marco satellite used three main components: a light outer shell; a heavy inner structure which contained the satellite's instrumentation and power supply; and the drag balance, an elastic system consisting of three orthogonal elements to link the two other

components. Because the outer shell was subjected to atmospheric drag, the three elements of the balance measured the displacement which was proportional to the force acting on the outer shell along three orthogonal axes. The experiment data were instantaneously transmitted to the ground when the satellite was over a receiving station because no recording instruments were on board.

2.2.2 Ionospheric Experiment

The ionospheric experiment investigated the ducting phenomenon in signal propagation (i.e., receiving radio signals beyond the normal receiving-station horizon), and measured the electron content of the atmosphere between the satellite and earth. For additional information on the ionospheric experiment, refer to the Centro Microonde publications.^{2,3,4}

2.3 LAUNCH VEHICLES

The San Marco project used three types of launch vehicles: Shotput and Nike-Apache rockets for suborbital flights in phase 1, and the Scout vehicle for orbital flights in phases 2 and 3. References 5 and 6 contain detailed descriptions of the Shotput and Scout vehicles.

2.4 TRAINING

The San Marco project was the first full-fledged space program instituted by Italian scientists. Because the Italian scientists planned to build a launch range and maintain a launch capability, the United States found it necessary to train Italian personnel for various functions. During 1963 and 1964, approximately 70 CRA engineers and technicians received training:

- GSFC trained CRA engineers in all facets of spacecraft design and operation, data-reduction systems, and dynamic balancing and testing.
- Langley Research Center (LRC) provided instructions on assembly and checkout of the Shotput vehicle.
- Ling-Temco-Vought (LTV), Scout vehicle contractor, held classes on assembly and checkout of the Scout vehicle.
- WI station provided information on range procedures and range-safety practices, and trained CRA personnel during NASA launch operations.

3. PHASE 1

The main objectives of phase 1 were to:

- Train Italian spacecraft engineers
- Train assembly and range-safety crews for Shotput and Scout vehicles
- Flight-qualify the air-density experiment and its associated subsystems
- Begin initial design of the equatorial launch complex

These objectives were attained between mid-1961 and mid-1963, by CRA engineers working with NASA counterparts to learn procedures, verify equipment design, and establish a working organization.

3.1 SHOTPUT 6 SPACECRAFT DESCRIPTION

The spacecraft (Figure 2) for Phase 1 Shotput vehicles was designed and fabricated in Rome. Parameters of the spacecraft⁷ were:

Size and shape:	26-inch-diameter sphere
Power supply:	series and parallel combinations of Mallory mercuric oxide RM-42R 1.3-v cells
Weight:	182 pounds
Battery weight:	60 pounds
Telemetry:	PAM/FM/PM; 136.53 MHz; 2-watt output IRIG channels 6, 7, and 8 (0 to 10 grams force) IRIG channels 3, 4, and 5 (0 to 100 grams force) IRIG 9 (commutator for housekeeping data)
Attitude:	unstabilized

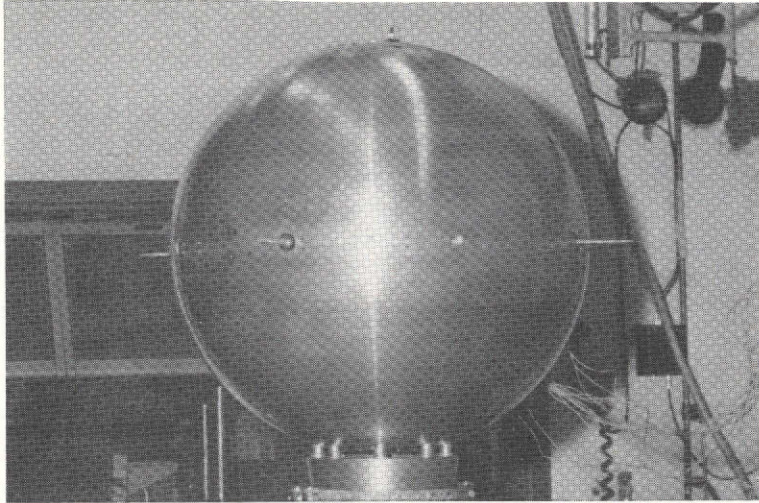


Figure 2. Shotput 6 Spacecraft (Phase 1)

The displacement of an element by an external force was converted to an electrical quantity by a differential transformer fed by a 2-kHz/second sinusoidal voltage. Each output was amplified, modulated, rectified, and made to modulate a voltage-controlled oscillator (VCO). The outputs were then summed by an integrating network whose output was fed to a phase-modulated transmitter.

The 160-inch-dipole antennas used for ionospheric data transmission were omitted from the Shotput spacecraft because the suborbital flight duration was too short for them to be extended. However, the transmitters for the ionospheric experiment were signal-tested on the Shotput spacecraft by monitoring their outputs on the housekeeping telemetry channels.

3.2 TESTING

By early 1963, CRA had established test facilities in Rome for acceleration, shock, and temperature tests. The Test and Evaluation Division (T&E), GSFC, performed the vibration and thermal-vacuum tests, and spacecraft balancing. GSFC also conducted complete structural tests and dynamic and static balancing of the engineering unit (a spacecraft model with all components except drag balance simulated), the prototype, and the first and second flight units. However, because of the heavy schedule of T&E, the task of vibration and thermal-vacuum testing was contracted to a private firm.

3.2.1 Spacecraft Balancing

San Marco was the first spacecraft to require or undergo triaxial balancing, the most complex precision balancing performed at GSFC to date. Because of the nature of the air-density experiment,⁷ forces not produced by air drag had to be minimized. To prevent spacecraft-induced forces while in the freedom of space, the spacecraft had to be dynamically balanced around the spin axis and around two perpendicular axes in the horizontal plane.

Before the San Marco project, spacecraft had been dynamically balanced only about their spin axes. The unique task of triaxially balancing a spacecraft was presented to T&E. The operation could not be performed in a step-by-step fashion, i.e., one axis at a time. The addition of weights in one plane to achieve balance around one axis could unbalance one or both of the other axes. Therefore, a different method⁸ had to be devised.

Problems were encountered in mounting the spacecraft on the balance machine because the original spacecraft design had no provision for this type of mounting. Threaded holes on the edges of the inner "wheel" section of the spacecraft defined the ends of the two perpendicular axes. Short stub shafts were attached to the holes so that the spacecraft could be mounted on the balance machine without excess weight which might cause undesirable inertia forces. The spacecraft was spun around each of the three axes, data were obtained on the amount of unbalance, and various analytical approaches were considered for obtaining balance.

A study was undertaken to establish, in mathematical terms, what was involved in the dynamic balancing of a spacecraft around each of its three perpendicular axes. Among other things, the study yielded two valuable conclusions which served to reduce the factors involved and to facilitate balancing operations. It was learned that when two axes were balanced, the third balanced automatically. It was also learned that balance weights must be placed in the centers of the edges of an imaginary cube, symmetrically surrounding the spacecraft, and that these points lay on the diagonal stiffeners in the spacecraft. Because the study showed that more weight was required to balance the spacecraft around three axes than to balance it around one, considerable research was conducted to minimize the amount of weight necessary.

After the San Marco prototype spacecraft was balanced, GSFC redesigned the balancing fixture for the flight units to allow spin about all three axes. Spacecraft handling was simplified, balancing sensitivity was improved and was approximately equal about all axes, and better alignment was eventually achieved. However, the increase in fixture size caused gravity-induced deflections,

requiring additional work on fixture design to increase its rigidity. Stiffening members attached to the fixture eliminated the problem and permitted better balancing than was possible before the redesign.

3.2.2 Qualification Tests

After static and dynamic balancing at GSFC, the prototype spacecraft was delivered to Aerotest for testing. Between January 27 and February 11, 1963, Aerotest conducted environmental qualification tests of the San Marco prototype spacecraft, consisting of vibration and thermal-vacuum exposures.

3.2.3 Acceptance Tests

After completion of qualification tests on the prototype, the first flight unit was statically and dynamically balanced on the redesigned fixture and shipped to Aerotest Laboratories for environmental acceptance tests. The test program conducted between March 19 and 27, 1963, included vibration, thermal-vacuum, and leak tests. Upon completion of the acceptance test program, the flight unit was ready for mating with the second-stage motor (X-248) of the Shotput vehicle.

3.3 SHOTPUT LAUNCH PREPARATION

The first flight unit was mated with the second stage of the Shotput 6 vehicle at Langley, and was delivered to Wallops Island Station for final spin-balancing on April 1963. The Tracking and Data Systems Division, GSFC, made preparations to provide data-recording support during launch to obtain experience with the San Marco telemetry signal and to supply CRA with data recorded in the GSFC format. The objective was to locate and resolve any interface problems between the GSFC acquisition system and the CRA processing system.

3.3.1 Launcher

The Shotput launcher (Figure 3) consisted of a vertical mast and boom, supported by a base and three support arms. The mast could be rotated 180 degrees on its longitudinal axis by an electric motor-driven tiller system. The boom was hinge-mounted to the base and could be raised or lowered by an electric motor-driven system of cables running between the boom and the top of the mast.

The launch vehicle was mounted to the mast, which was positioned to establish the azimuth angle. Electromechanical indicator systems (one at the hinge of the boom, the other at the upper end of the mast) provided azimuth and zenith angles to indicators in the blockhouse.

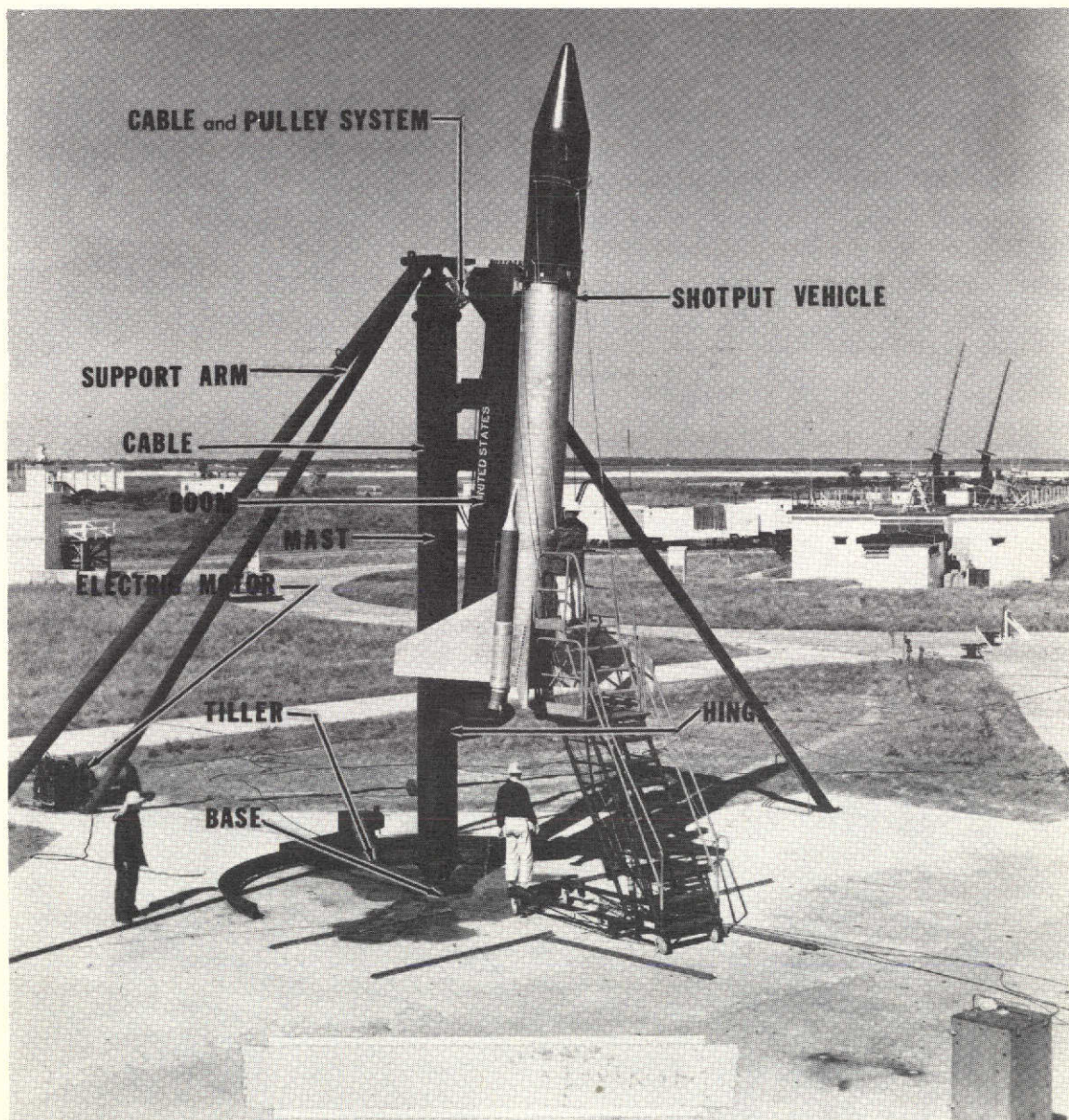


Figure 3. Shotput Launcher

3.3.2 Vehicle

The Shotput vehicles (Figure 4) used for the suborbital San Marco launches were two-stage rockets spin-stabilized by skewed aerodynamic fins. The first stage was a Pollux E6 motor with two recruit motors strapped on. The second stage was an X-248 motor.

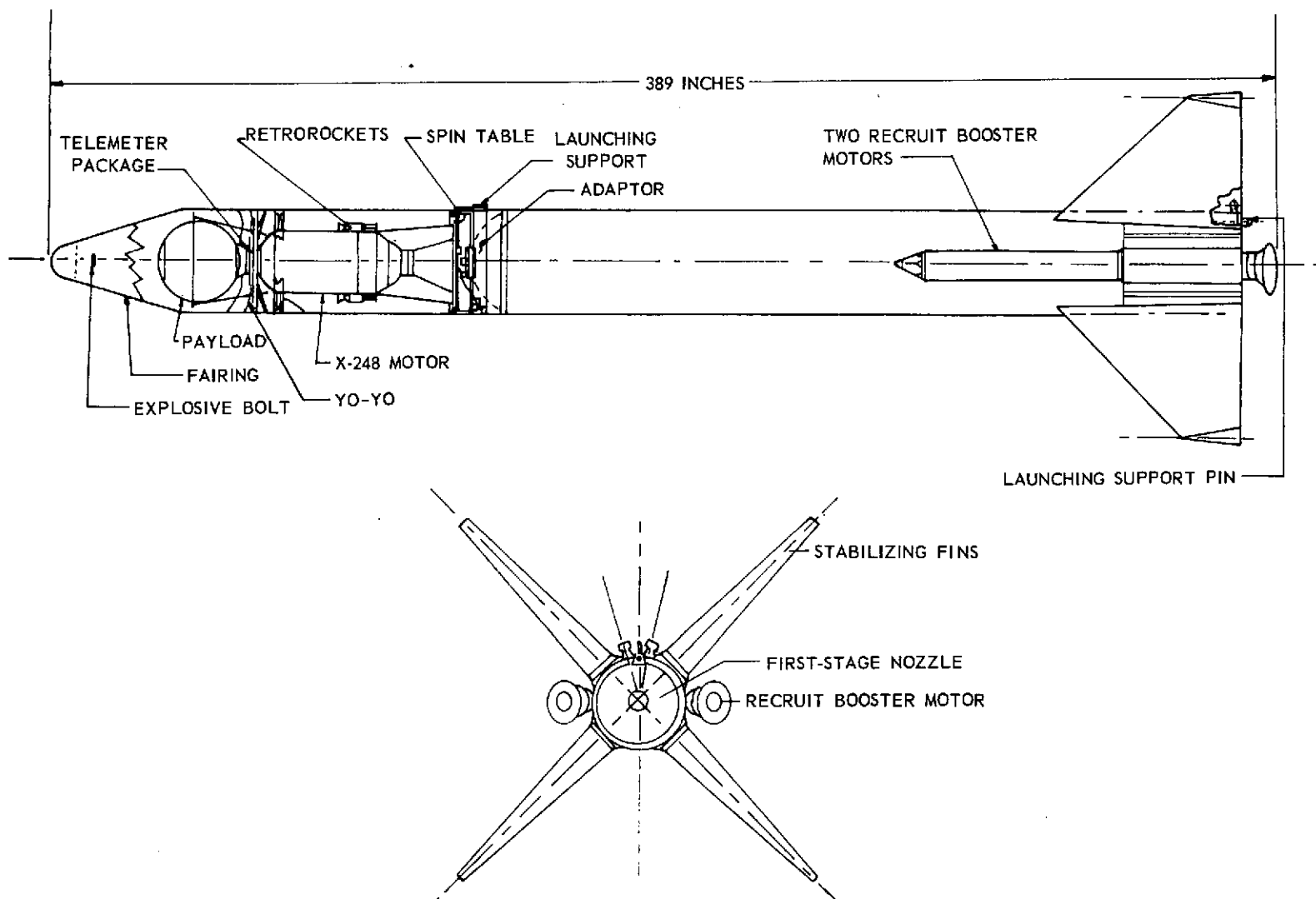


Figure 4. Shotput Vehicle

The vehicles were modified for the San Marco launches by locking the spin tables (normally used for despinning the second-stage and spacecraft assembly from 260 rpm to 160 rpm before second-stage ignition). This change allowed the second-stage/spacecraft assembly to be fired while spinning at about 260 rpm. After second-stage burnout and before second-stage separation, the yo-yo despin system was sized to reduce the spin rate of the assembly to less than 6 rpm. Separation of the spacecraft from the second stage was accomplished by means of a mechanical spring and, at a later time, augmented with the firing of six retro-rockets located around the periphery of the second stage.

3.4 SHOTPUT 6 LAUNCH

3.4.1 Launch

Flight unit 1 was launched April 21, 1963, on Shotput 6, from WI (Figure 5). After the first-stage burnout and coast phase, the heat shield was ejected, the explosive bolts holding the second stage to the spin table were fired, and the second stage was ignited.



Figure 5. Shotput 6 Launch

Upon second-stage burnout, the despin system failed to reduce the spin rate of the second-stage/spacecraft combination.* The high spin rate persisted after second-stage/spacecraft separation, making atmospheric measurements with the drag balance impossible. The vehicle also attained a higher trajectory than predicted (240 nautical miles instead of 195 nautical miles), causing the spacecraft to impact outside the expected area.

3.4.2 Results

Except for the yo-yo despin system, all other spacecraft systems functioned properly. Despite the failure in obtaining experiment-data results, the flight was considered to be successful because valuable data were obtained on the associated subsystems.

NASA and CRA agreed to conduct a second Shotput launch from W1.

3.5 SHOTPUT 7 SPACECRAFT DESCRIPTION

The Shotput 7 spacecraft (flight-unit 2) weighed 177.5 pounds. It differed from the Shotput 6 spacecraft primarily in balance sensitivity. The low-force channels were made more sensitive⁹ with a range of 0 to 6 grams. The high-force channels remained at 0 to 100 grams. In addition, CRA engineers incorporated a completely redundant caging system; this characteristic had been previously recommended and stressed by GSFC.

3.6 TESTING

The T&E Division conducted the flight-unit-2 acceptance tests, including:

- Vibration test (June 5 and 6, 1963)
- Thermal-vacuum test (June 10 and 11, 1963)
- Dynamic balance (June 13, 1963)

*The definite cause of failure was not established at a meeting held between LRC and GSFC representatives. However, the condition strongly suggested the nonrelease of one of the two yo-yo weights. It was agreed that such a malfunction could have been induced by either of the following conditions: (1) a screw on a release-pin assembly may have vibrated loose; (2) the power supply may not have been adequate to actuate the four dimple motors in parallel.

- 360-rpm 5-minute-duration spin test (June 13, 1963)
- 2-minute-flight random (thrust axis only) vibration test (June 13, 1963)
- Mass measurements (June 14, 1963)

After environmental tests, the spacecraft was taken to Beltsville where the Sounding Rocket Instrumentation Section conducted a complete telemetry check. The flight unit was then shipped to LRC for mating with the second stage and for interface checks with the Shotput vehicle. Finally, the total assembly was shipped to WI where the spacecraft/X-248 combination was spin-balanced.

3.7 SHOTPUT 7 LAUNCH

3.7.1 Launch

The Shotput 7 spacecraft (flight-unit 2) was successfully launched from WI on August 2, 1963. The Shotput vehicle lifted the payload to an altitude of 155 nautical miles, covering a surface distance of 560 nautical miles.

Despin from a maximum of 335 rpm to less than 1 rpm, payload separation, balance uncaging, and balance damping all occurred properly; telemetry from the spacecraft was normal.

3.7.2 Results

The three high-force (0- to 100-gram scale) channel readings deviated slightly. Two of the three low-force (0- to 6-gram scale) channel readings went off scale. Upon re-entry into denser atmosphere, both readings came back on scale for a time, then went off scale in the opposite direction as the force exceeded their range. This problem was attributed to a probable "permanent set" of the balances caused by 2 months of continual caging.

Telemetry of performance parameters indicated that all sequences of events were normal and that the vehicle performed within established tolerances. The principal discrepancy in vehicle performance was a large yaw dispersion. The trajectory also was somewhat low at payload separation.

On the up leg of the flight, the first test interval lasted from 129.4 to 180 seconds; the second lasted from 8 minutes 20 seconds to 9 minutes 20 seconds on the down leg of the flight.

Figure 6 shows the predicted trajectory for the Shotput launches. Shotput 7 achieved the predicted trajectory.

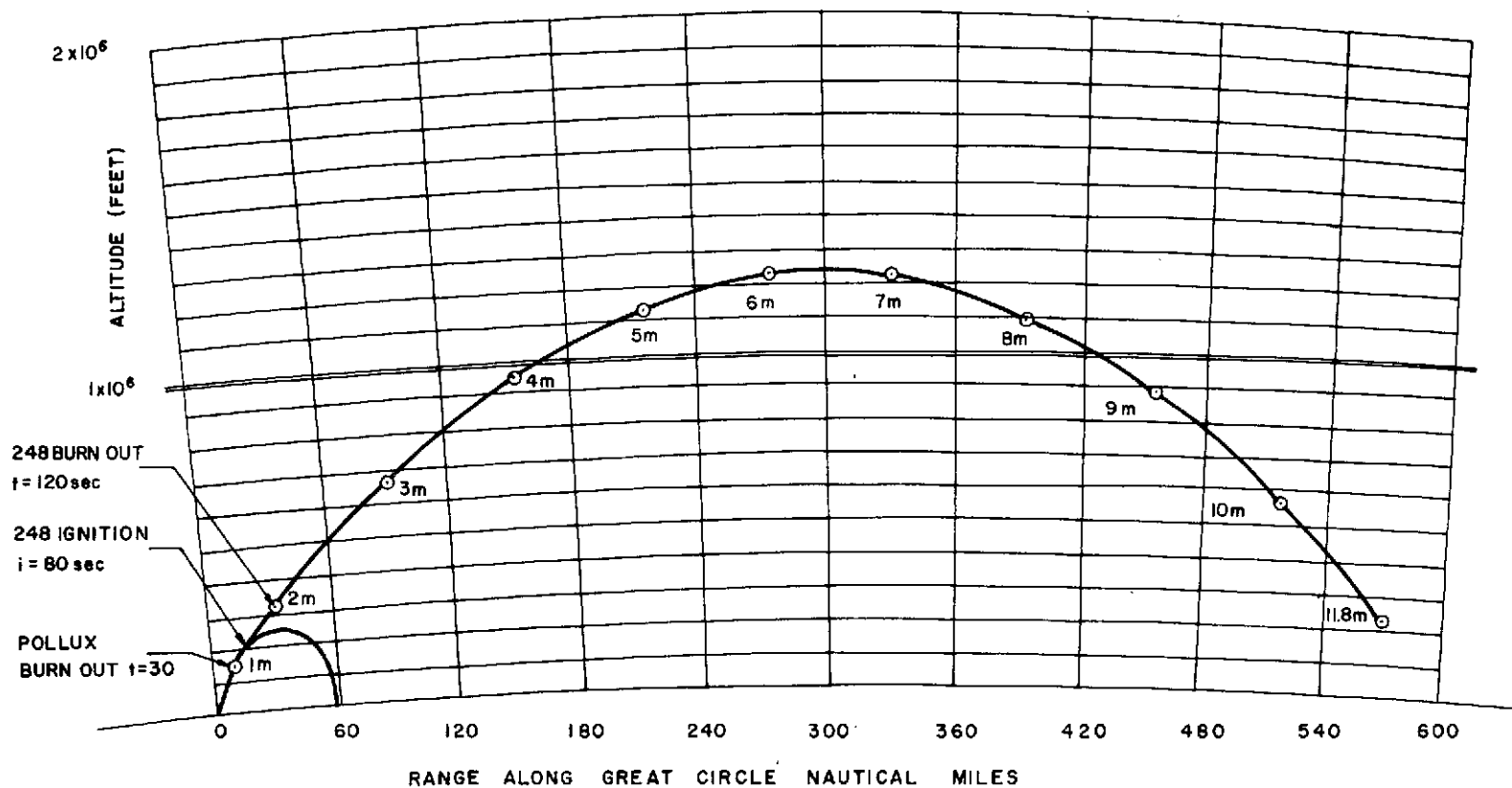


Figure 6. Predicted Trajectory for Shotput Launches

3.8 SHOTPUT 8 OPERATIONS

By mutual agreement, the concluding operations of phase 1 were amended to include the launch of a third spacecraft using a Shotput vehicle from the Santa Rita platform at the equatorial launch site in Kenya. The primary purpose of the Shotput 8 platform launch was to validate the launch range with a suborbital operation before using the Scout vehicle. Except for the more complex logistics necessary for the Scout, many operations would be similar for both the orbital and suborbital programs.

3.9 SHOTPUT 8 SPACECRAFT DESCRIPTION

The Shotput 8 spacecraft systems¹⁰ were similar to previous systems with one exception: Instead of having separate high and low (force) telemetry channels, each of the three force balances now had an automatic gain level selection requiring only half the number of telemetry channels.

3.10 TESTS AT THE ROME FACILITY

The following arrangements were made to provide test and evaluation assistance requested by CRA:

1. The Rome test facility drawings submitted by CRA were evaluated.
2. GSFC San Marco staff personnel (vibration engineer, T&E test manager, telemetry engineer, and project coordinator) were sent to Rome as advisors during the October 1963 tests.
3. Preparations made for environmental testing included:
 - Resolution of spacecraft operating modes
 - Establishment of test procedures
 - Integration, electrical checkout, potting, and re-check of the subsystems in the prototype unit.
 - A Goddard-built subcarrier-oscillator test device to measure bandwidth versus noise ratio was incorporated by CRA in their ground-telemetry checkout system.
 - The thermal-vacuum facility, originally an 11-foot-diameter chamber with a nitrogen shroud, was modified to include a 4-foot-diameter CO₂ shroud.

The test plan for the spacecraft and operating procedures for the environmental test equipment and ground station were prepared by CRA engineers in cooperation with GSFC representatives. Prototype spacecraft checkout was completed November 6, 1963; preliminary static balance was completed November 7; and random vibration tests on the thrust axis were completed November 15.

Subsequent acceptance tests¹¹ on the flight model were completed within 1 week.

Because the balancing equipment did not arrive on time, the flight spacecraft had to be shipped to GSFC for balancing. Triaxial-dynamic balance and moments-of-inertia measurements were completed January 8, 1964, at Goddard. The spacecraft was then shipped to LRC on January 16, 1964, for fit checks. After the fit checks were completed, the spacecraft was shipped to WI and mated with the second stage, and the combination was spin-balanced. The spacecraft telemetry was checked out and exercised at WI before the spacecraft/second-stage combination was crated for shipment to the launch complex at Kenya.

The spacecraft/second-stage combination was scheduled to leave the United States on January 28, 1964; however, because an early monsoon season was predicted at the launch site, CRA requested a slip in the launch schedule from mid-March to the fall of 1964. All equipment except the spacecraft was unpacked and prepared for storage at WI; the spacecraft was shipped to Rome for checkout and several modifications.

3.11 KENYA RANGE VALIDATION

Meanwhile, CRA scheduled a series of Nike-Apache launches from the Santa Rita platform and decided that the Shotgun 8 launch objectives could be achieved by substituting a Nike-Apache vehicle. One launch was planned for checking the compatibility of ground-station equipment with telemetry equipment on the rocket (similar to the telemetry equipment on the Shotgun 8 payload). The GSFC Sounding Rocket Branch provided hardware and technical assistance for installing telemetry equipment in the Nike-Apache vehicle. Some circuit changes were required to match the 136-MHz San Marco telemetry to the 240-MHz equipment normally used on the Nike-Apache. The equipment was then shipped to Rome for further wiring and assembly; after checkout, it was shipped to the Santa Rita platform for installation in the rocket. The Nike-Apache was launched March 26, 1964, concluding phase 1.

4. PHASE 2

Primary objectives of phase 2 were to:

- Train crews for Scout vehicle assembly, and for checkout, launch, and range operations
- Flight-qualify the air-density and ionospheric experiments, and the spacecraft subsystems under actual orbital conditions
- Obtain data on air-density and ionospheric characteristics

4.1 SAN MARCO A SPACECRAFT DESCRIPTION

4.1.1 Structure

The San Marco A satellite (Figure 7) was spherical for aerodynamic isotropy; protruding parts were kept to a minimum. Its internal structure provided both

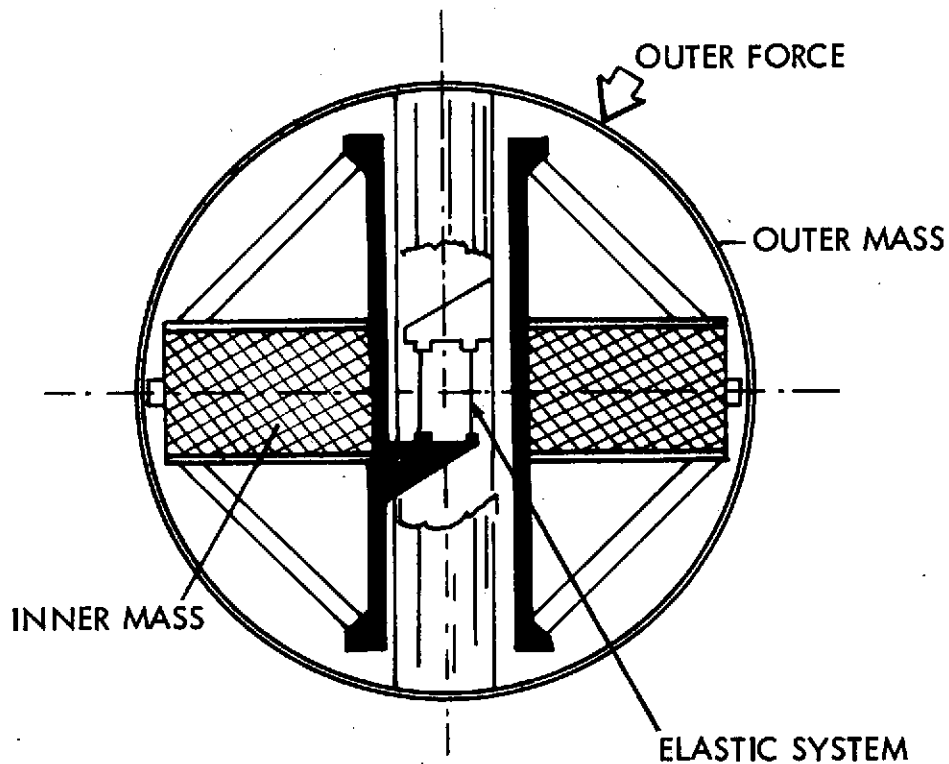


Figure 7. Internal Layout of San Marco A

mechanical strength and thermal paths for conducting heat away from the delicate instrumentation. The balance at the center of the inner member was protected against excessive launch stress by a pneumatic caging system which connected the external shell to the internal structure. Elements of the balance were separately caged during launch by pneumatic actuators.

The shell design was modified (Figure 8) to open along a longitudinal axis so that it could be removed and components inspected without removing the spacecraft from the last stage of the rocket motor. This modification eliminated the need to rebalance the spacecraft/last-stage assembly after inspection.

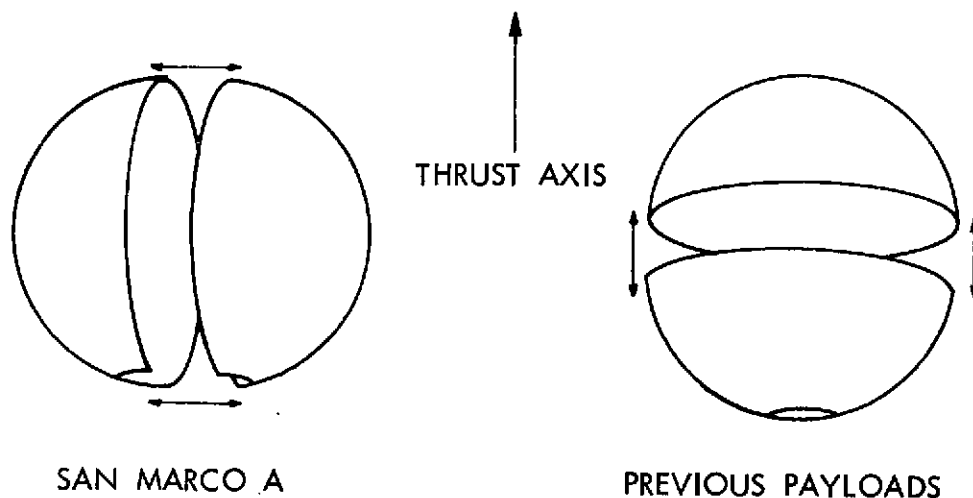


Figure 8. Modified Shell Design

4.1.2 Atmospheric-Drag Experiment

The San Marco A atmospheric-drag experiment was modified to use the same IRIG channels (through automatic switching) for both the low-force and the high-force scale. Sensitivity of the low-force scale was increased from 0 to 6, to 0 to 2.5 grams, and that of the high-force scale from 0 to 100, to 0 to 25 grams. When a force greater than 2.5 grams was encountered, the range (amplifier gain) was automatically switched to the 0- to 25-gram scale. The three axes were vectorially summed and fed into a fourth channel which modulated the beacon transmitter. A fifth telemetry channel was used for sending house-keeping data. The spacecraft was also modified to permit all three balances to be individually zeroed by ground command when the spacecraft was out of the earth's atmosphere (at apogee).

The drag balance is a 3-degree-of-freedom mechanical system in which the degrees of freedom are represented by linear displacements. Because each displacement is obtained by means of a single element (Figure 9), there are three elements, each connected to the other in series. Linear displacement arises from the elasticity (normal to their planes) of the thin plates (springs) fitted at the ends in two rigid bases which represent the reference and the moving

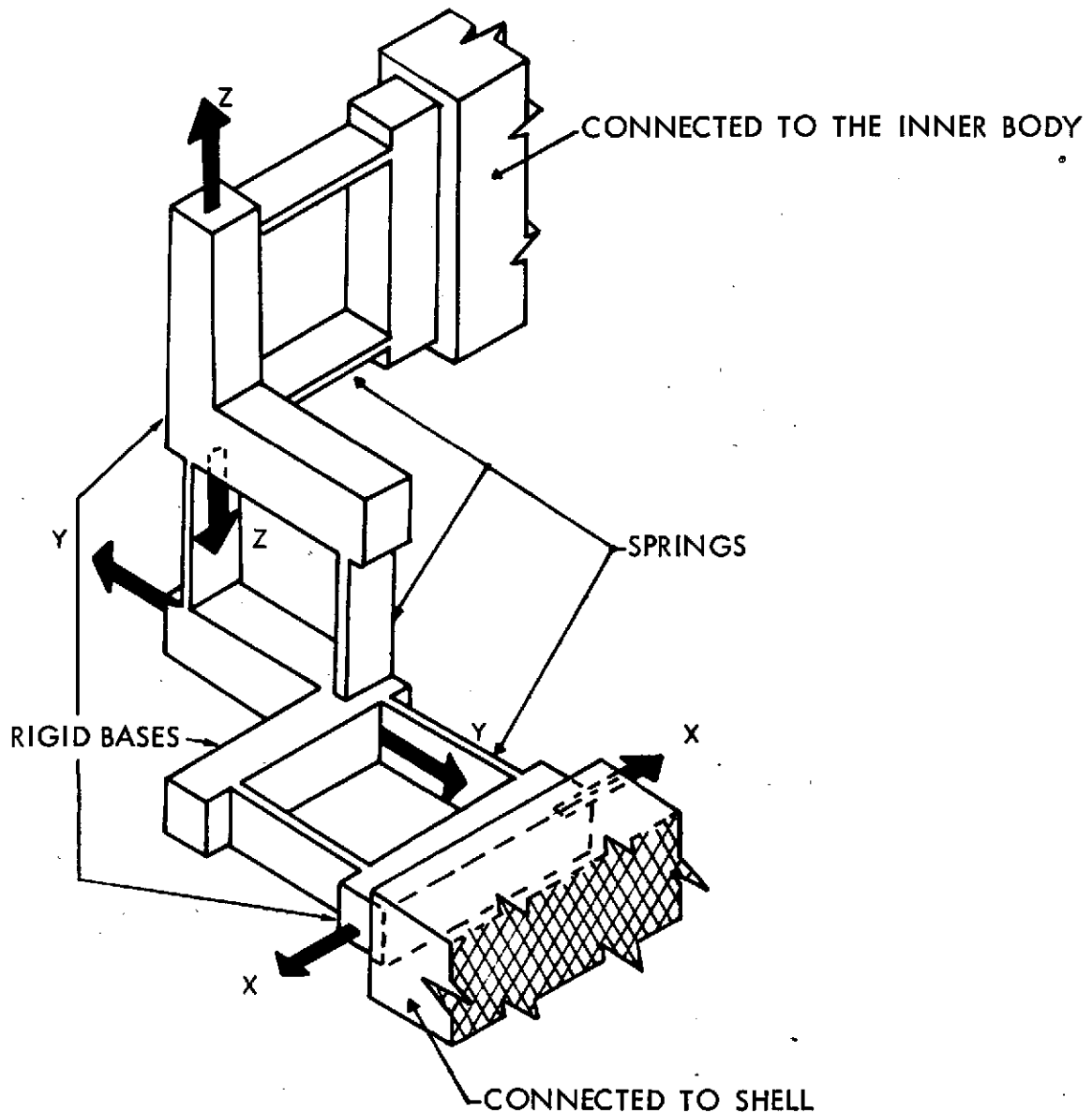


Figure 9. Drag-Balance Schematic

point, respectively. Therefore, the single element has a preferential direction of primary motion caused by the bending of the springs. Electric transducers (differential transformers with coils and core on the fixed base and on the moving base, respectively) transform this motion into an electrical signal. Because of the linear elasticity of the system, the signal is assumed to be a proportional measurement of the force.

Each element of the mechanical system is centered to avoid moments on the balance, as well as for satellite dynamics. Reference 12 contains further information.

4.1.3 Ionospheric Experiment

- The ionospheric experiment used a high-frequency (20.005 MHz) 670-milliwatt transmitter which was frequency-shift-keyed (FSK) with a 100-Hz shift to facilitate recognition of the radiated signal. The transmitter output was fed through matching networks to the two monopoles of an extensible 160-inch-dipole antenna. The antenna was extended only when the ionospheric experiment was turned on by ground command.

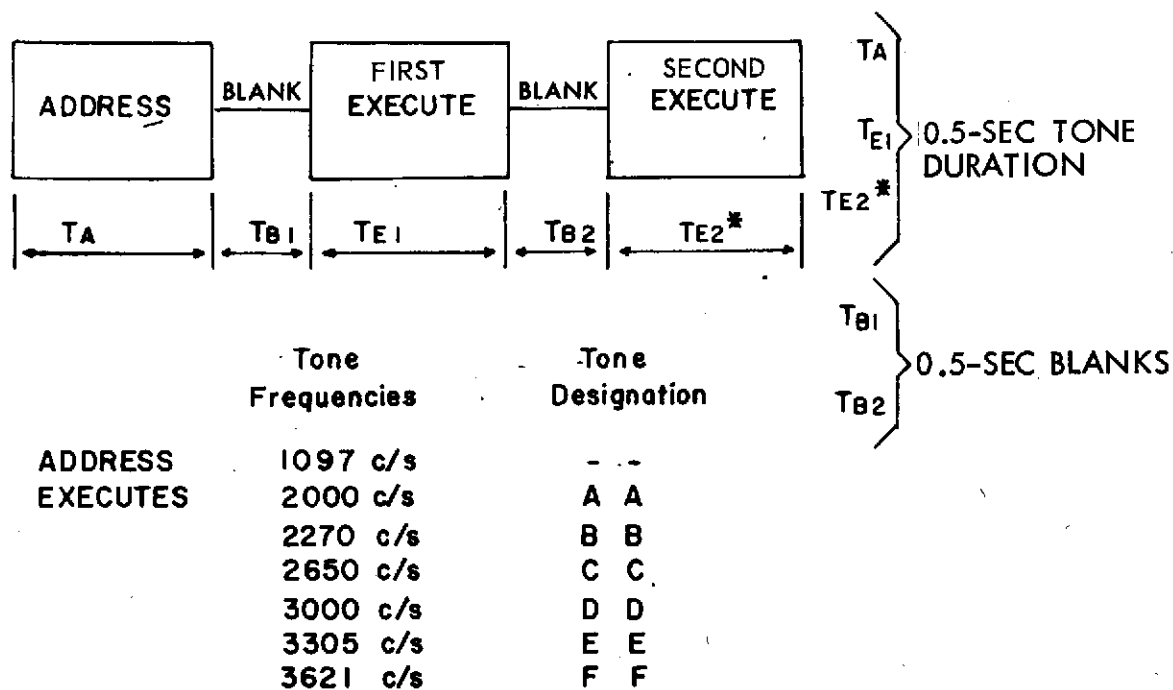
4.1.4 Telemetry System

The pulse-amplitude-modulated/frequency-modulated/phase-modulated (PAM/FM/PM) system was used for telemetry. The spacecraft used four continuous-transmission channels for the atmospheric-drag experiment and one sub-commutated channel for housekeeping signals. The ionospheric experiment housekeeping data were included in the subcommutated channel. The transmitter was crystal-controlled, 0.25-watt output, and phase-modulated.

4.1.5 Command System

Ground commands were sent to the spacecraft through a radio-frequency (RF) link on the 140- to 150-MHz band using a standard audio-command system (with sequential tone transmission). An address tone sent to arm the tone decoder was followed by two execute tones to accomplish a command function. Figure 10 shows the ground-command frame format and tone sequences. (Goddard tone-command standards were followed.)

The spacecraft command system consisted of a command receiver, a tone decoder, and a command decoder. The receiver detected the RF ground command to extract the audio which was fed to the tone decoder where the tone signals were rectified. The separate dc outputs were fed to the command decoder which decoded the two execute tone combinations sent from the ground. The address tone armed the



* T_{E2} is variable from 0.05 to 1.0 sec for the CRA station only.

Figure 10. Command Frame Format

decoder for 3 seconds; execute tones therefore, had to be received within this time to accomplish the desired function.

4.1.6 Timer

A timer circuit was included to turn the air-density experiment on if a turn-on command was not received within 24 hours during the orbital life of the spacecraft. The timer circuit was provided to circumvent a failure of the command link.

4.1.7 Power Supply

Mercury batteries with a nominal life of 200 working hours provided power for the satellite. The basic Mallory RM-42R cell was used. The maximum instantaneous power level did not exceed 12 watts.

4.1.8 Beacon Transmitter

A 50-milliwatt-beacon transmitter was used for tracking. The vector sum of the three balance channels was used to modulate the 136.74-MHz carrier frequency.

4.2 TESTING

Both prototype and flight units for the San Marco A launch were environmentally tested in Rome at the CRA facility, but were dynamically balanced at GSFC. The prototype was integrated and electrically tested early in August 1964. GSFC project staff members reviewed the spacecraft subsystem designs and discussed their findings with CRA personnel.

Tests conducted on San Marco A were:

<u>Prototype Unit</u>	<u>Flight Unit</u>
Static and dynamic balance	Balance
Temperature test	Vibration
Vibration	Spin
Acceleration	Leak test
Spin	Solar simulation
Leak test	Thermal vacuum
Solar simulation	Balance refinement
Operation and deployment tests	Physical measurements
Thermal vacuum	
Physical measurements (center-of-gravity, weight, moment-of-inertia)	

LTV, Dallas, Texas, conducted the mechanical fit, radio-frequency interference (RFI), and heat-shield ejection tests (Figures 11 and 12) in October 1964, using the prototype spacecraft.

A compatibility test conducted at Blossom Point on November 4, 1964, with the same prototype spacecraft, indicated that all tracking, telemetry, and command links between the San Marco spacecraft and STADAN were compatible.

Meanwhile, the flight spacecraft arrived at GSFC from Rome on October 20, 1964 for dynamic balancing; the flight-separation system arrived from LRC on October 28, 1964, for weight, center-of-gravity, and moment-of-inertia measurements. Final spacecraft weight was 249.5 pounds, including 1.24 pounds of balance weights.* The separation system, including the yo-yo despin system, weighed 22.2 pounds.

*The shell, which could not be assembled to the spacecraft until the balancing procedure was performed, weighed 6.50 pounds.

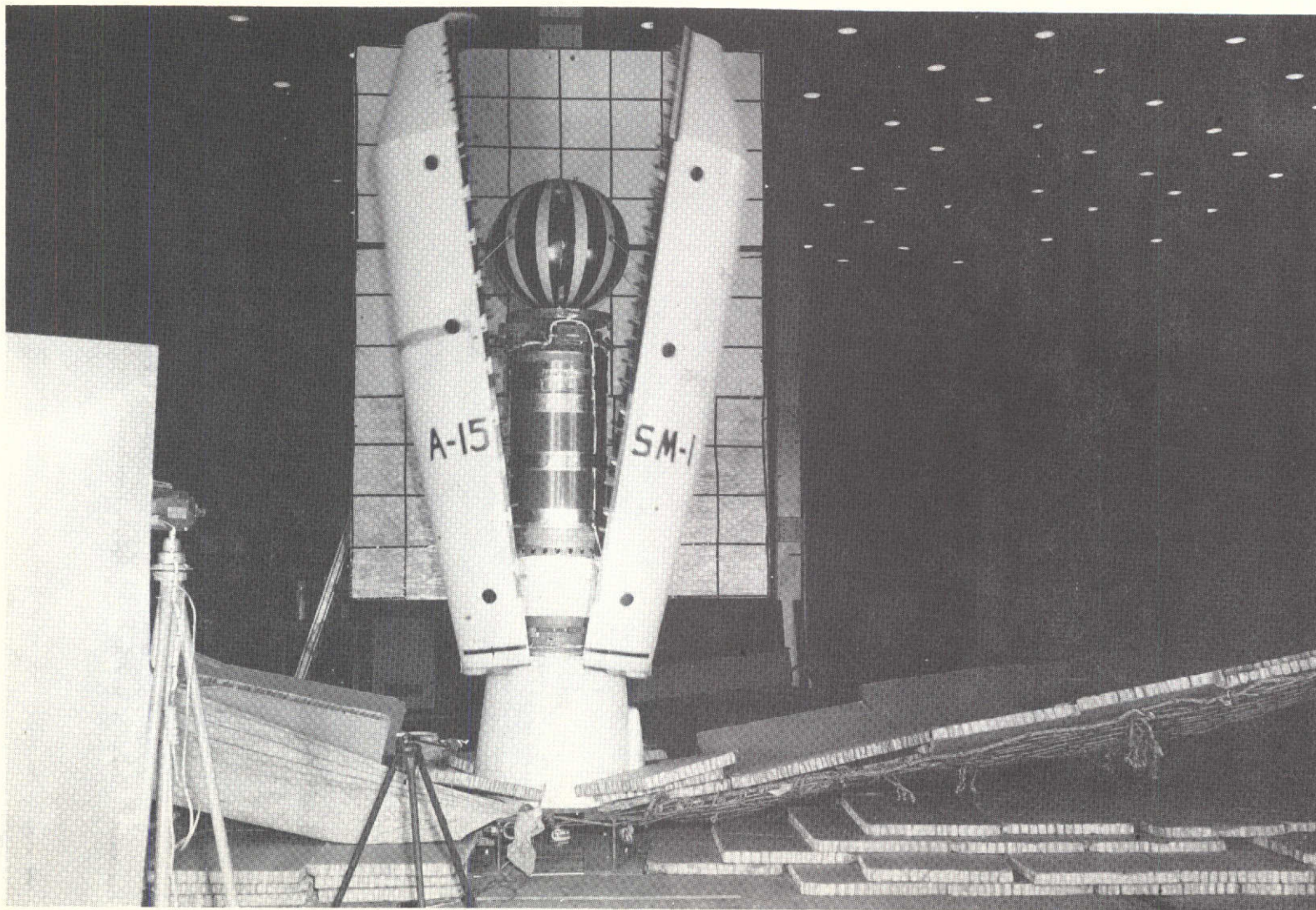


Figure 11. Scout A-15 Heat-Shield Fit and Ejection Test With San Marco A

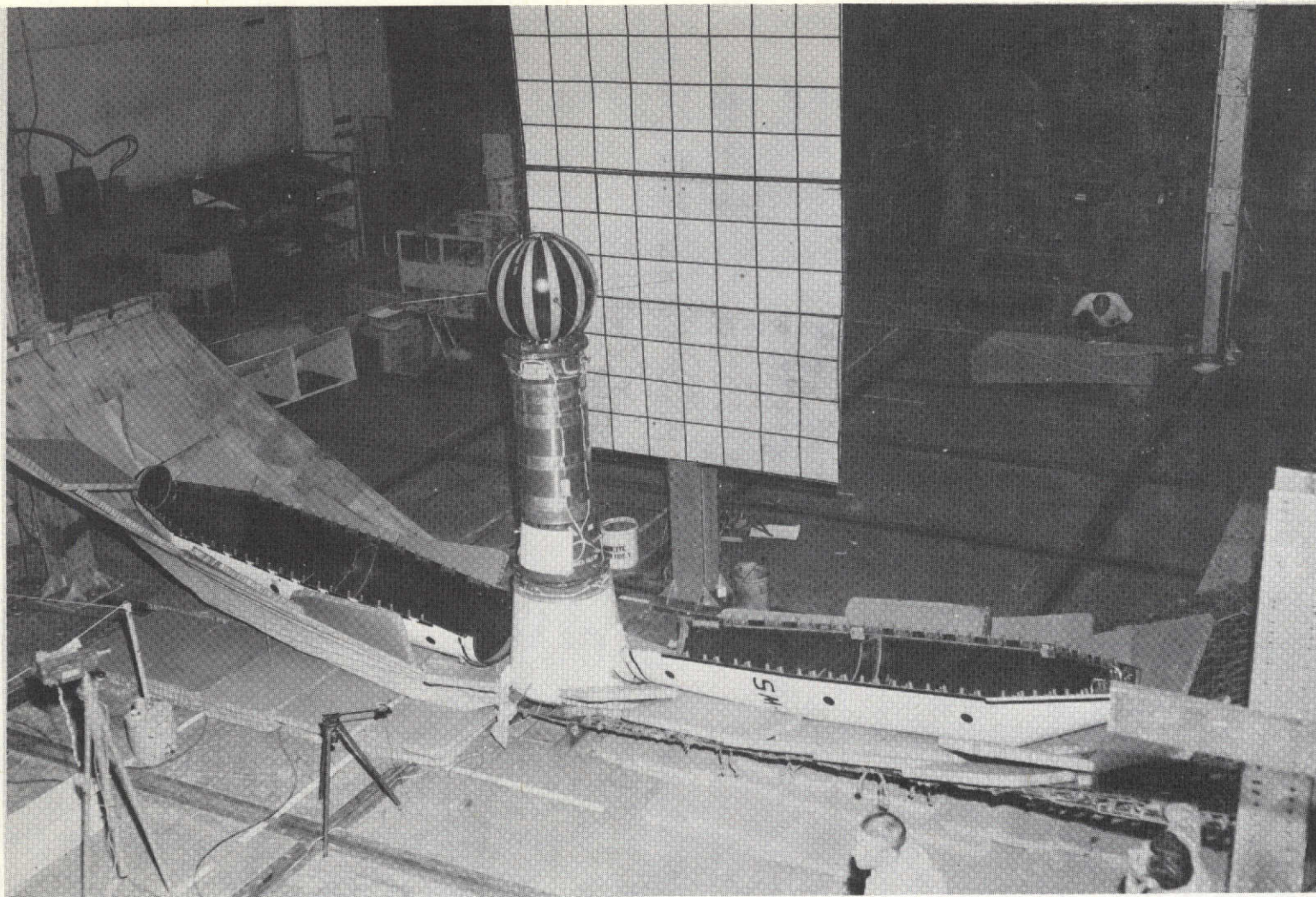


Figure 12. Scout A-15 Heat-Shield Fit and Ejection Test With San Marco A

On November 17, 1964, the combination of flight spacecraft and Scout fourth stage was spin-balanced and assembled with the third stage at WI.

4.3 MARK II SCOUT LAUNCHER

Mark II Scout launcher (Figure 13) consists primarily of a tower structure to which the vehicle is attached and a base which supports, erects, and rotates the structure. The vehicle is attached to this structure by two sets of retractable support arms and two support ring-pins (Figure 14). The combination can be set in any position from horizontal to vertical, permitting vehicle inspection in the horizontal position. The base is set on a turntable capable of positioning the structure at any launch azimuth from 0 to 180 degrees. The tower structure, in its horizontal position, is covered by a shelter which is supported on a wheel-and-track assembly, and which is rolled back when the tower is raised. Duplicate sets of controls and indicators on the base of the launcher and in the blockhouse permit azimuth and elevation changes from either location. The support arms and the umbilical flyaways operate pneumatically, the support arms on compressed air and the umbilicals on nitrogen.

A typical sequence of events for launcher operation is:

- The vehicle is positioned under the launcher by a transporter.
- The launcher tower is lowered to the vehicle.
- The vehicle is attached to the structure by two pairs of retractable support arms (at the top and center of the vehicle) and two launch support ring-pins (at the center and base of the vehicle).
- The shelter is rolled back and the transporter is removed.
- The tower is raised and rotated to position the vehicle for launch (Figures 13 and 14).
- Before firing, the support arms are retracted and the umbilicals are released; the ring-pins release at liftoff and spread to clear the vehicle fins.

The San Marco A launch was the third project to use the Mark II launch facility at WI.

4.4 SCOUT VEHICLE

The Scout vehicle is a four-stage-propellant rocket.^{6,13} San Marco A Scout (Figure 15) was 72 feet long and weighed 20 tons at liftoff.

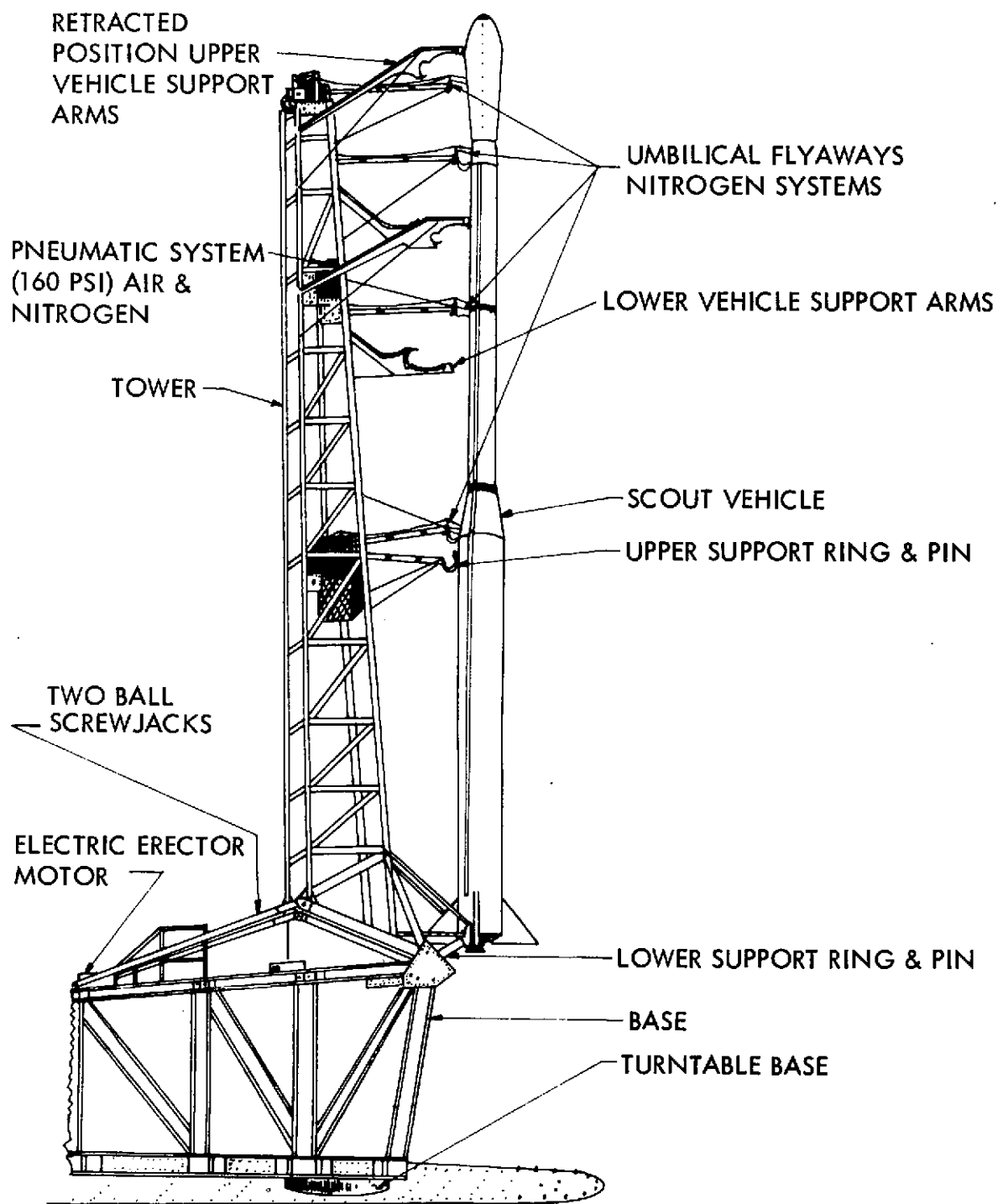


Figure 13. Mark II Scout Launcher

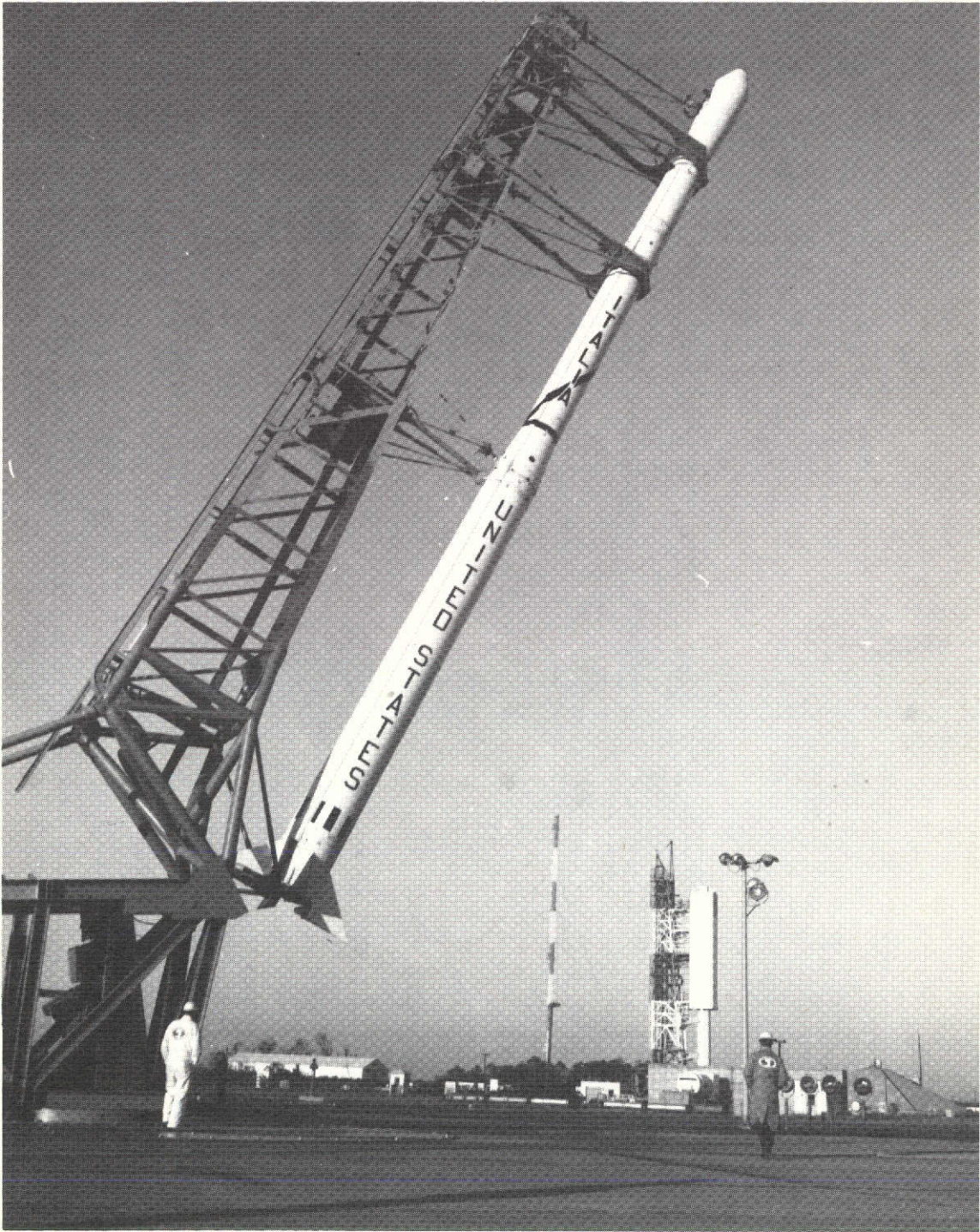


Figure 14. Launcher Operation

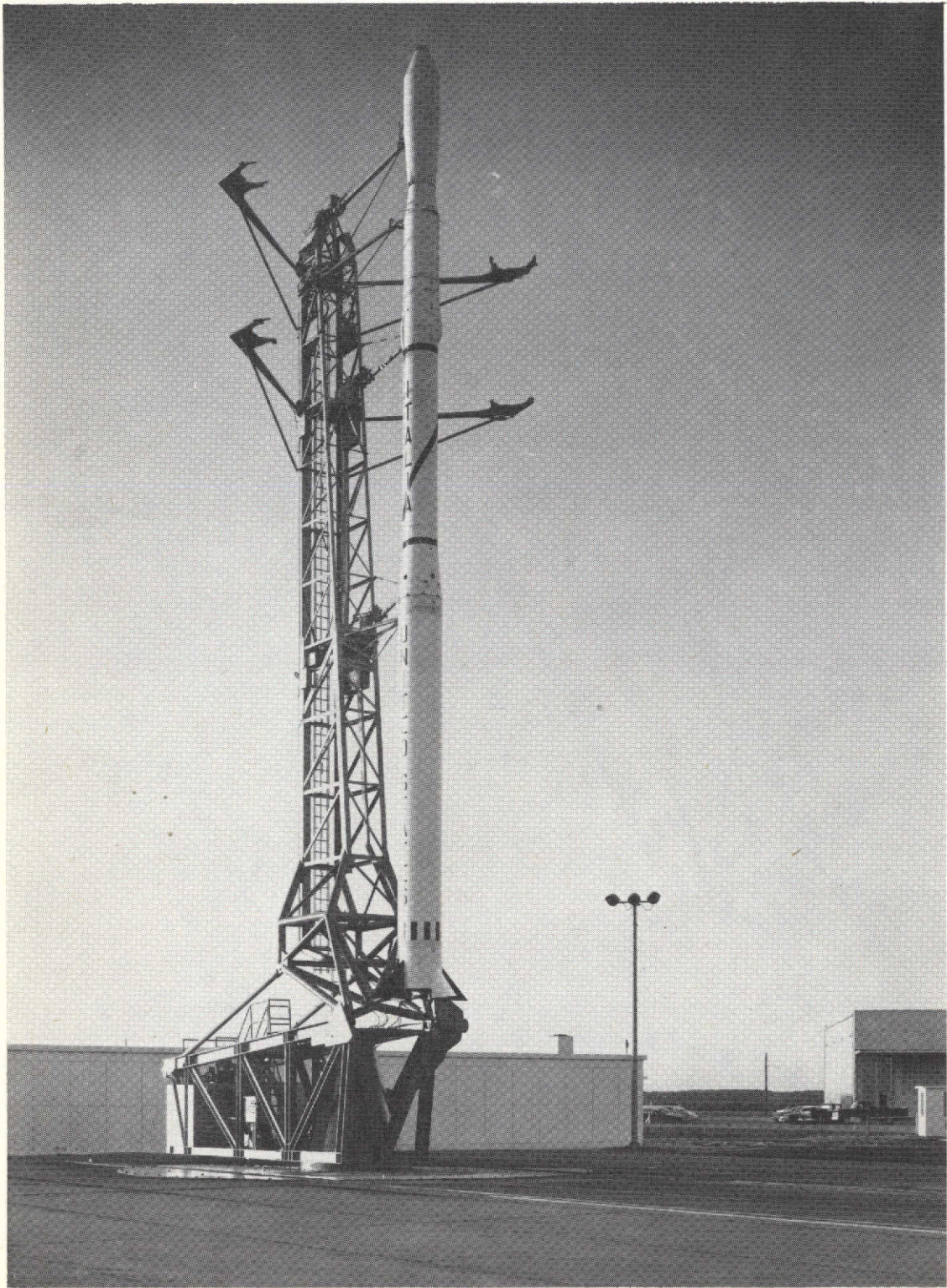


Figure 15. San Marco A Scout

The four stages for the San Marco A launch were:

- First stage, Algol 11B
- Second stage, Castor I
- Third stage, Antares II (ABLX-259)
- Fourth stage, Altair (ABLX-258)

Figure 16 shows the assembled spacecraft and fourth-stage motor complete with heat shield.

4.5 SAN MARCO A LAUNCH

The San Marco A spacecraft was launched by the CRA crew from WI (Figure 17) into an elliptical orbit (205-km perigee, 820-km apogee, 37.8-degree inclination) on December 15, 1964. The launch was normal. Table 1 lists the nominal and actual orbital elements.

Table 1

Orbital Elements for San Marco A

Element	Nominal	Actual
Inclination	37.69 degrees	37.79 degrees
Period	90 minutes	95 minutes
Spin rate	0 to 6 rpm	3 rpm (measured from AGC)
Height of perigee	215 km	206 km
Height of apogee	680 km	820 km

Figure 18 shows injection and typical orbits of the satellite.

4.6 SPACECRAFT PERFORMANCE

Some difficulty was encountered obtaining quick-look data during and immediately following injection; however, the on-board sequences were established by the first orbit. For the next few days, the satellite did not respond to all commands. Investigation indicated that the satellite had a limited receiving range, and that

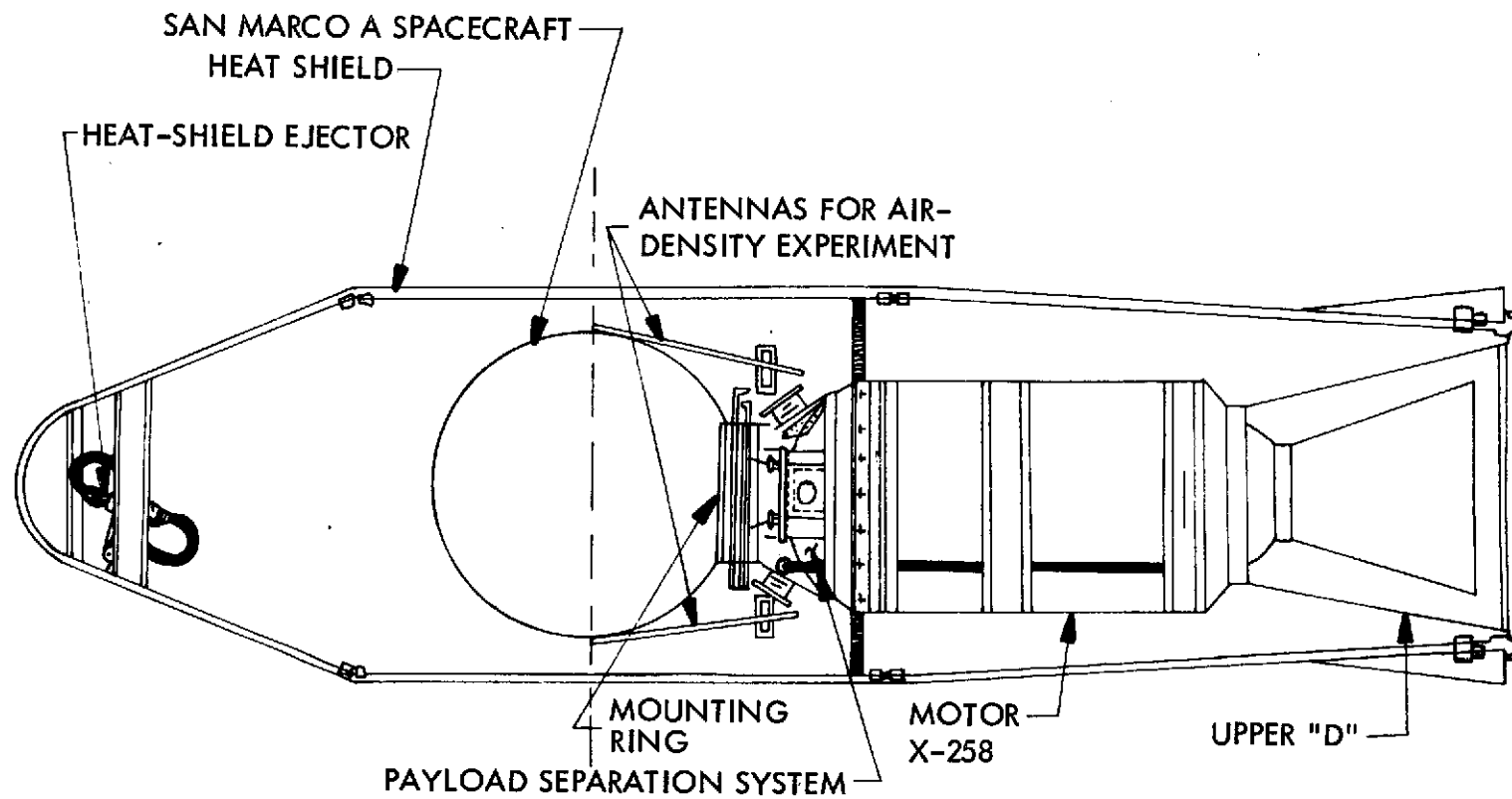


Figure 16. San Marco A Spacecraft/Fourth-Stage Motor Assembly

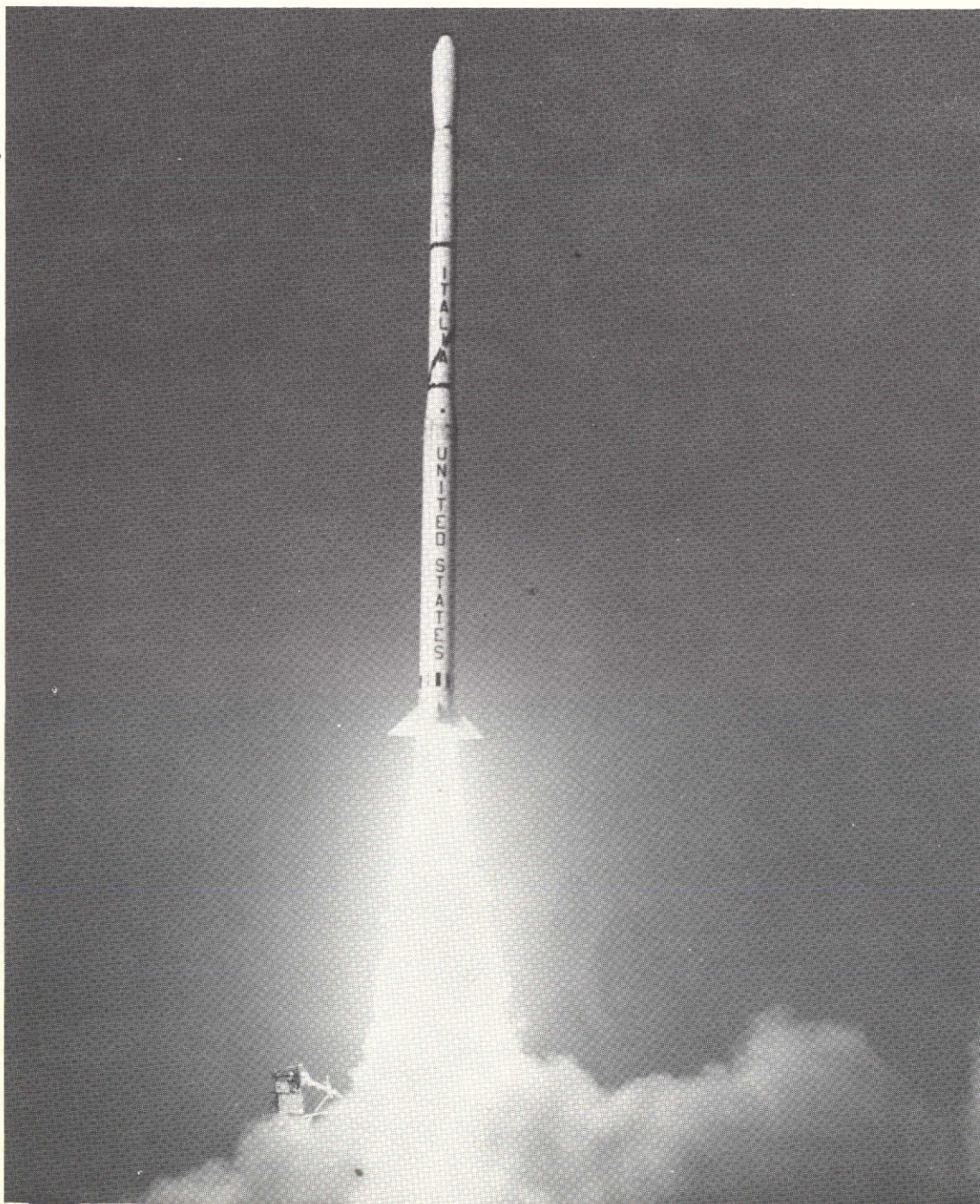
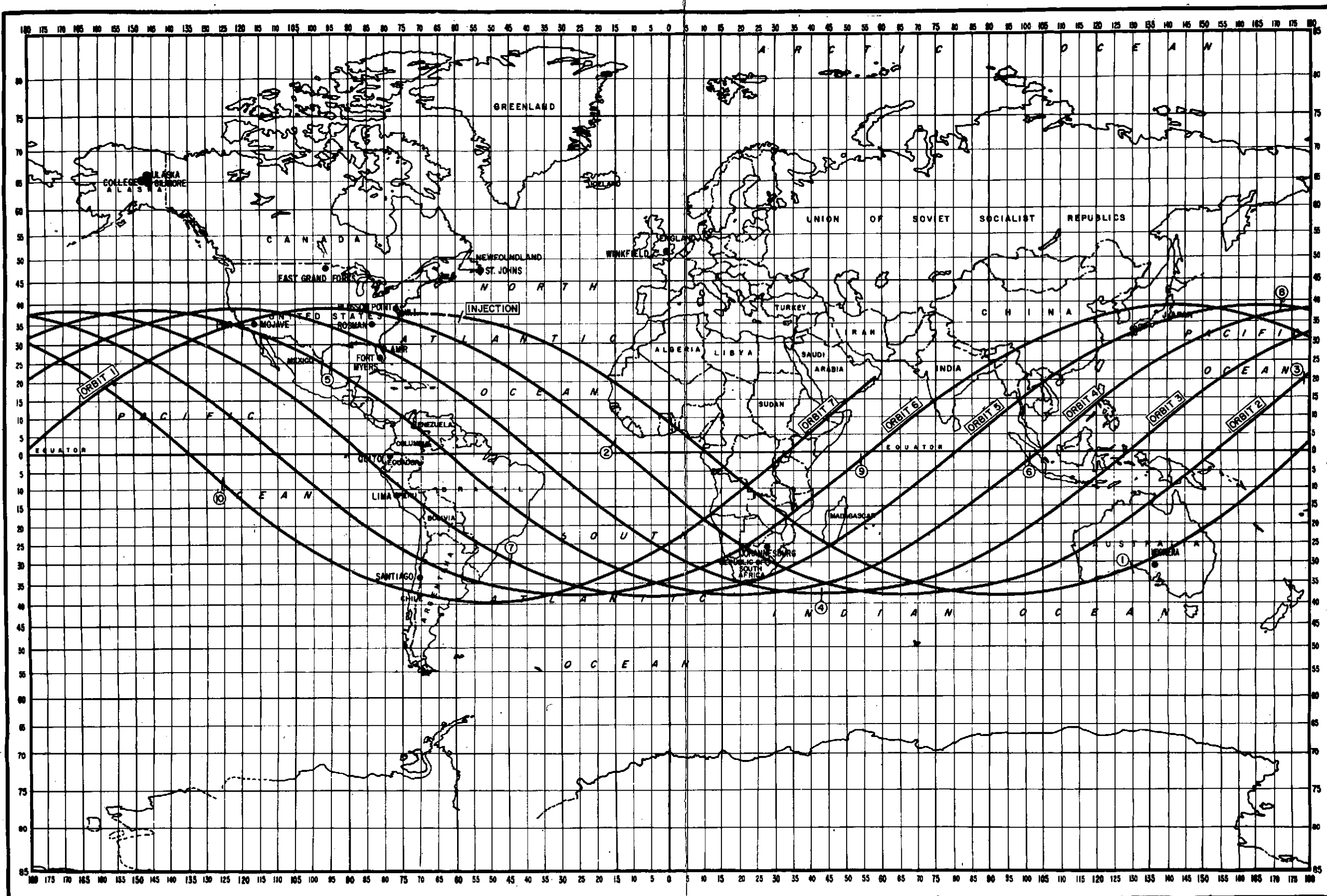


Figure 17. San Marco A Launch



NOTE: NUMBERED TIC MARKS INDICATE
HOURS AFTER LAUNCH.

FOLDOUT FRAME

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Figure 18. Nominal Orbit Subsatellite Plot, San Marco A

FOLDOUT FRAME

judicious choice of satellite/ground-station distance improved command response. The drag-balance channels operated as expected; voltages and temperatures were normal.

The on-board calibration steps were normal except for a double calibration which one tracking station noted after the fifth day. At times the satellite could not be turned off because it was out of range of the station or because of a higher priority pass at the ground station. This problem was significant because of the limited (200-hour) operational lifetime of the satellite's power supply, and because of a circuit that had been incorporated to circumvent a command receiver failure (if one occurred) by turning the spacecraft telemetry on permanently, 24 hours after the spacecraft received the last on or off command.

After data had been acquired for several weeks, the voltage which supplied the housekeeping data reference signal, the in-flight calibration circuit, and the force-channel demodulator was approaching the lower limit for operation. The decreasing voltage was thought to be caused by an open diode in one of the two parallel strings of cells.

As a power-saving measure, all long-term transmissions from the spacecraft were stopped. On December 30, 1964, CRA requested that all air-drag experiment transmissions be terminated in order to activate the ionospheric experiment. After acquiring data for the scheduled period of about 2 weeks, the experimenters requested that the ionospheric experiment be turned off. It was calculated that most of the separate ionospheric experiment power had been consumed. Because of its complexity, the command procedure for this operation provided GSFC and CRA personnel with much valuable experience.

A perigee plot showed that, during the lifetime of the experiment, the spacecraft travelled through the atmospheric-density bulge which was then located in the southern hemisphere.

4.7 TRACKING AND DATA-SYSTEM FUNCTIONS

The Tracking and Data Systems Division, GSFC, provided the following support for the San Marco A launch:

4.7.1 Training

A CRA engineer learned the GSFC data-processing system, specifically (with the aid of GSFC personnel) to prepare functional specifications for a CRA data-processing system.

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The specifications had to be kept general because the capabilities of the Italian manufacturers who were to provide the system were not known. The basic requirements (format, speed, etc., rather than definite configurations) were supplied and the Italian firm, to which the contract was awarded, worked out the actual configuration. The configuration, which the Italian firm furnished and which GSFC personnel reviewed and considered suitable, provided growth potential, allowing CRA to expand its usefulness by feeding additional programs to the system instead of buying new equipment.

4.7.2 Orbit Determination

As part of the prelaunch operations for San Marco A, primary orbit determination was computed to provide the Minitrack stations with look-angles so that they could locate and command the satellite, and to provide the experimenter with the precise location of the satellite when data were taken.

A number of orbit configurations was established which met the requirements for both the ionospheric and air-density experiments; a number of final orbit configurations with varying decay times was submitted to CRA. The final choice was an orbit¹⁴ which would keep the perigee at approximately 200 km for an orbital lifetime of 6 months.

4.7.3 Predicted World Maps

After the orbit was selected, a predicted world map was generated showing the nominal orbit for the first 3 days after launch. The map provided various tracking stations with nominal look angles, times, and elevations for the first 24 hours after launch while early tracking data were integrated into the program.

A production world map and station predictions were generated from the first determined orbit. Suborbital paths were projected on the world map, showing satellite longitude, latitude, and altitude. Station predictions consist of information calculated for each station, including slant-range, azimuth, and elevation correlated to time in 1-minute intervals. Scheduling normally requires updating of the map each week; however, updating was required every 3 days for San Marco 1 to compensate for early arrival of the satellite over tracking stations.

4.7.4 Operations Plan

The operations plan¹⁵ integrates spacecraft/experiment requirements with STADAN capabilities to supply information to STADAN on how to track, command, and acquire telemetry from the San Marco 1 satellite.

4.7.5 Data Acquisition

STADAN stations acquired data from the air-density experiment eight or 10 times a day between launch (December 15, 1964) and at the end of telemetry operation (February 15, 1965). These data were recorded on analog magnetic tapes correlated to Greenwich mean time (GMT). The tapes were then shipped to GSFC for quality checking and then to Rome for data processing and merging with orbit data by the air-density experimenters.

The ionospheric experiment transmitter (20.005 MHz) was turned on for 2 weeks, about 2 weeks after launch. During the 2-week operating period, ionospheric experiment receivers at the tracking stations were turned on continuously. The received data were recorded on continuously operating time-correlated magnetic tape recorders. The ducting phenomenon was detected a number of times, particularly on tapes from the station at Woomera, Australia. The signal from the ionospheric experiment transmitter was also recorded on stripcharts to measure the total electron content in the ionosphere when the spacecraft was overhead.

Magnetic tapes and stripcharts with data from the ionospheric experiments were sent from the tracking stations to GSFC for quality checking; from GSFC, tapes and charts were sent to the Centro Microonde at Florence, the headquarters for ionospheric studies in the San Marco project. Ionospheric data were then correlated with orbital data by the ionospheric experimenters.

4.7.6 Orbit Computation

After the satellite stopped transmitting data in mid-February 1965, final satellite ephemerides were calculated for San Marco 1, showing the satellite position in space and in time with 1-minute increments for the 2-month period during which the spacecraft transmitted data to earth. The computation of ephemerides required a considerable amount of time because CRA had requested unusually close tolerances: a 1-km-radius sphere at perigee, and a 2-km-radius sphere at apogee. To calculate the ephemerides, the air-drag effects had to be considered. Preliminary comparisons of ephemerides air-drag data with CRA air-drag data from the on-board experiment showed GSFC calculations to be accurate within 10 to 15 percent.

4.7.7 Spacecraft Control

The Space Physics Operations Control Center at GSFC was in charge of spacecraft control for San Marco 1. Quick-look data received from Minitrack stations three or four times a day were used to generate housekeeping stripcharts showing changes in temperature, battery power, and component conditions on board

the satellite. Because the 24-hour safety timer on San Marco 1 proved to be temperature-dependent, it was necessary to determine when the timer would go on in order to know when to turn it off. This proved to be more difficult than had been expected, because the safety-timer operation was influenced by temperature in such a way that the time-on period varied from approximately 22.5 to approximately 24.5 hours. With a total power capacity of only about 2 weeks on the San Marco A spacecraft, the 1.5-hour variations in power consumption presented a loss too great to be permitted. The timing period had to be recalculated (considering temperature) each day in order to know when to command the timer off.

4.7.8 Data Reduction

CRA was responsible for processing data from San Marco 1. Magnetic tapes from the tracking stations were quality-checked at GSFC and sent to CRA for reduction. The University of Rome processed the air-density experiment data; the Centro Microonde processed the ionospheric data. The GSFC Control Center received stripcharts from the stations and, after analysis, sent them to Rome for preliminary reduction of the scientific data. (CRA's automatic data-reduction facility was not operational when the San Marco A was launched.) The CRA analysis of spacecraft motion indicated that the spacecraft, spinning at 1 rpm, experienced a precession of 6 rpm; this condition complicated data acquisition and data analysis.

Altogether, 1861 minutes of air-density data and 2068 minutes of ionospheric data were acquired.

4.8 RESULTS

4.8.1 Mission

NASA units (WI, LRC, and GSFC) which participated in the San Marco project met in January 1965, to review the objectives of phase 2 and to assess the degree to which each objective had been achieved.

Each Center agreed that the objectives in its area were more than adequately met and that CRA had maintained a high standard of excellence in all of its endeavors. The consensus was that the success of the mission formally ended phase 2.

CRA and NASA subsequently agreed to proceed with phase 3 of the program.

4.8.2 Scientific

CRA presented, in a preliminary paper, the scientific data obtained from the air-density experiment¹⁶ to the Committee on Space Research (COSPAR) meeting in Buenos Aires in May 1965. CRA presented the final results¹⁷ to the COSPAR meeting in Vienna in May 1966.

4.8.3 Spacecraft Design Philosophy

GSFC analyzed the spacecraft performance and recommended a number of changes.¹⁸ Specific recommendations from GSFC to CRA for consideration in future spacecraft were:

1. Power supply: Use solar cells, rechargeable batteries, and associated circuitry to increase power-supply life commensurate with orbit life.
2. Safety timer: If solar cells are not used, increase the time period of the timer and change the operation to recycle the timer with every on or off command.
3. Tape recorder: Incorporate a small lightweight tape recorder to provide data storage and fast playback to enable full-orbit data acquisition. (Low-orbit satellites have limited real-time-telemetry contact with ground stations because the satellite is visible over each station for only a short time and the number of stations visible per orbit is small.)
4. Quick-look data: Use an interranger instrumentation group (IRIG) format commutator on the spacecraft, and automatic-decommutation equipment in at least one ground station to permit housekeeping data to be reduced quickly and accurately without going through the normal data link.
5. Command sequences: Simplify command sequences by adding more commands or by modifying the circuits for less complex command sequences.
6. Circuits: Regulate the voltage of the reference (or sync) signal or the commutator to at least the minimum under which the system will operate; check transmitter preemphasis setup to determine why some telemetry channels had more noise than others when transmitting low-RF signals; and improve command-receiver sensitivity and eliminate level shift.

The phase 3 spacecraft (San Marco B) remained similar to the phase 2 spacecraft because there was not enough time between phases to allow the incorporation of all of these recommendations into the program. However, most of these recommendations were included in the San Marco C program, which followed phase 3.

4.9 REENTRY

North American Radar Defense (NORAD) observed the San Marco 1 (A) reentry which occurred at 1100 universal time (Z) on September 14, 1965, at an estimated location of 34°N, 173°W. Later, it was reported that a part of a pole piece of the fourth-stage igniter, identified as part of vehicle 137R, was found in Thailand.

5. PHASE 3

Final plans for phase 3 were completed during 1965. CRA contracted with LTV for a Scout Mark II launcher and associated mechanical ground-support equipment (GSE). CRA built the electrical GSE from drawings and specifications supplied by LTV.

5.1 OBJECTIVES

The objectives of phase 3 were twofold: to investigate the atmosphere in the equatorial region, and to make the equatorial launch site operational.

5.2 SAN MARCO B SPACECRAFT DESCRIPTION

CRA personnel in Rome designed, fabricated, integrated, and tested the San Marco B prototype and flight spacecraft.¹⁹ GSFC contributed consultative assistance. LTV conducted fit checks and radio-interference tests in Dallas with the prototype spacecraft. Later, GSFC conducted STADAN-compatibility tests with the same spacecraft.

The basic design of San Marco B and San Marco A was the same; however, the circuitry on B was redesigned to increase reliability and to reduce power consumption, thereby increasing satellite operating life. In addition, the sensitivity of the air-drag experiment on San Marco B was improved.

Drag-balance sensitivity selections and zero-shift corrections (within ± 80 percent) were made by ground command. (Ten milligrams on the 0- to 1-gram scale corresponded roughly to 350-km altitude; 25 grams on the 0- to 25-gram scale corresponded to approximately 120-km altitude.)

The telemetry system (Figure 19) remained a PAM/FM/PM system with four continuous channels and one subcommutated channel; however, a circuit was added to turn the telemetry off automatically after 8 minutes of transmission if the telemetry was not turned off by the interrogating ground station.

From a previous recommendation, the timer and control circuit (Figure 20) was modified to provide an override to the command system in case of failure (i.e., if the spacecraft could not be interrogated in 6 days). Table 2 lists the 6-day cycle provided when the circuit is actuated.

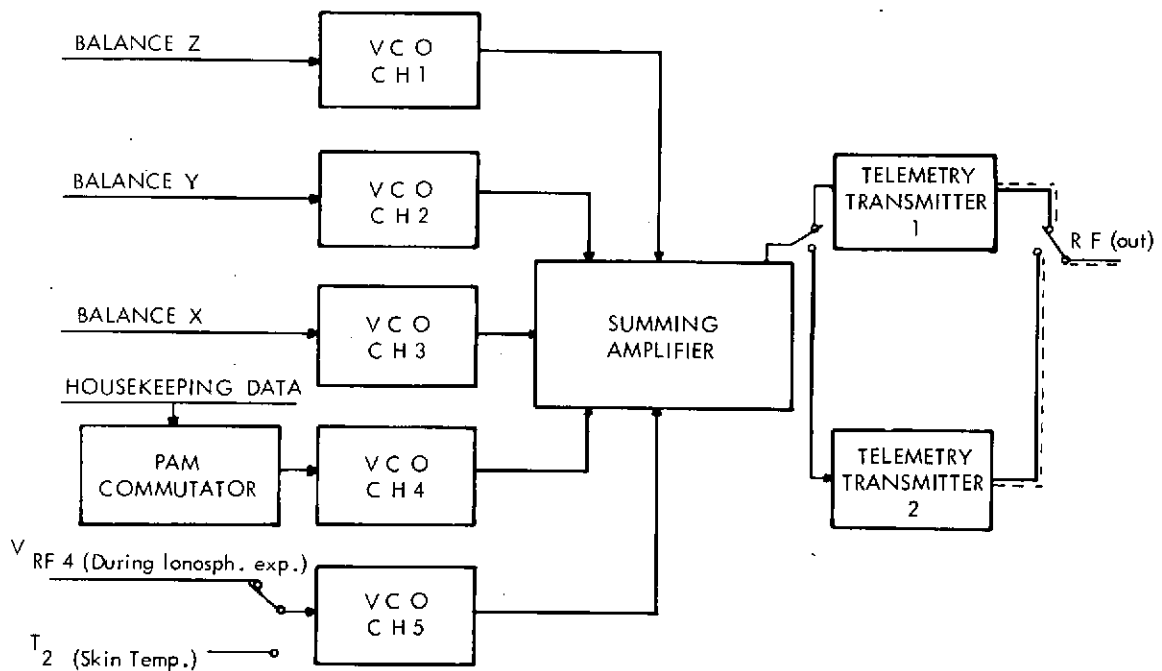


Figure 19. Telemetry System Block Diagram

Table 2

6-Day Timing Cycle

Period (days)	Air-Density Experiment and Telemetry	Ionospheric Experiment
1.2	On	Off
4.8	Off	On

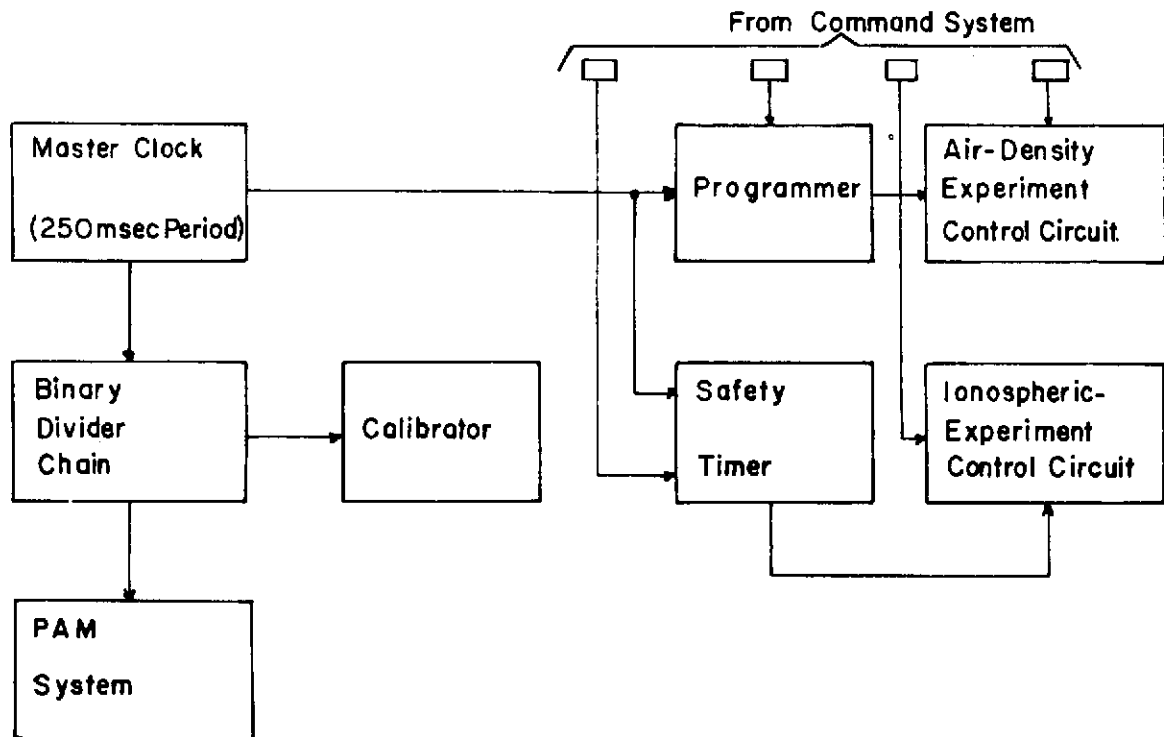


Figure 20. Timer and Control-Circuit Block Diagram

If the command system functioned normally, the safety timer was reset by each turnon-turnoff command and therefore remained inactive.

5.3 KENYA LAUNCH COMPLEX

The equatorial launch site was located in the Formosa Bay in the Indian Ocean, approximately 90 miles north of Mombasa, Kenya. The platforms (Figure 21) were located approximately 3 miles off the coast near the village of Ngomeni, at a point on the continental shelf where the water is about 30 feet deep. Both platforms were raised so that the bottom decks stood about 15 feet from the highest known water level. A small plot of land was leased from the Kenya government for a shore-based camp where additional personnel were accommodated in tents and open buildings, i.e., poured concrete pads with a roof. Personnel were transported in open boats from a pier at the camp site to the offshore platforms.

5.3.1 Santa Rita Platform

The Santa Rita platform (Figure 22) is a triangular platform (115-foot sides with a 15-foot hull depth) similar to those used for offshore drilling rigs, commonly

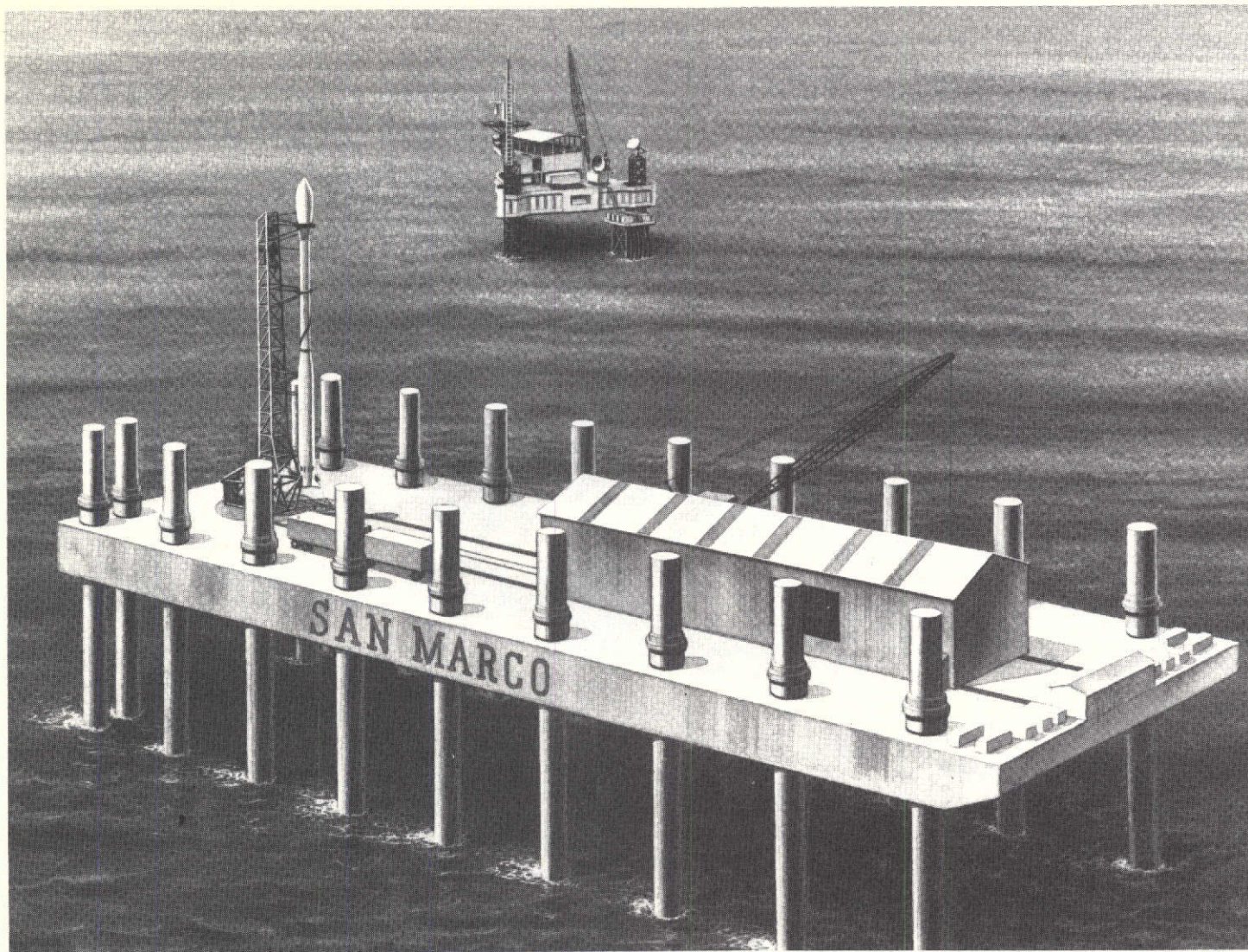


Figure 21. Kenya Launch Complex

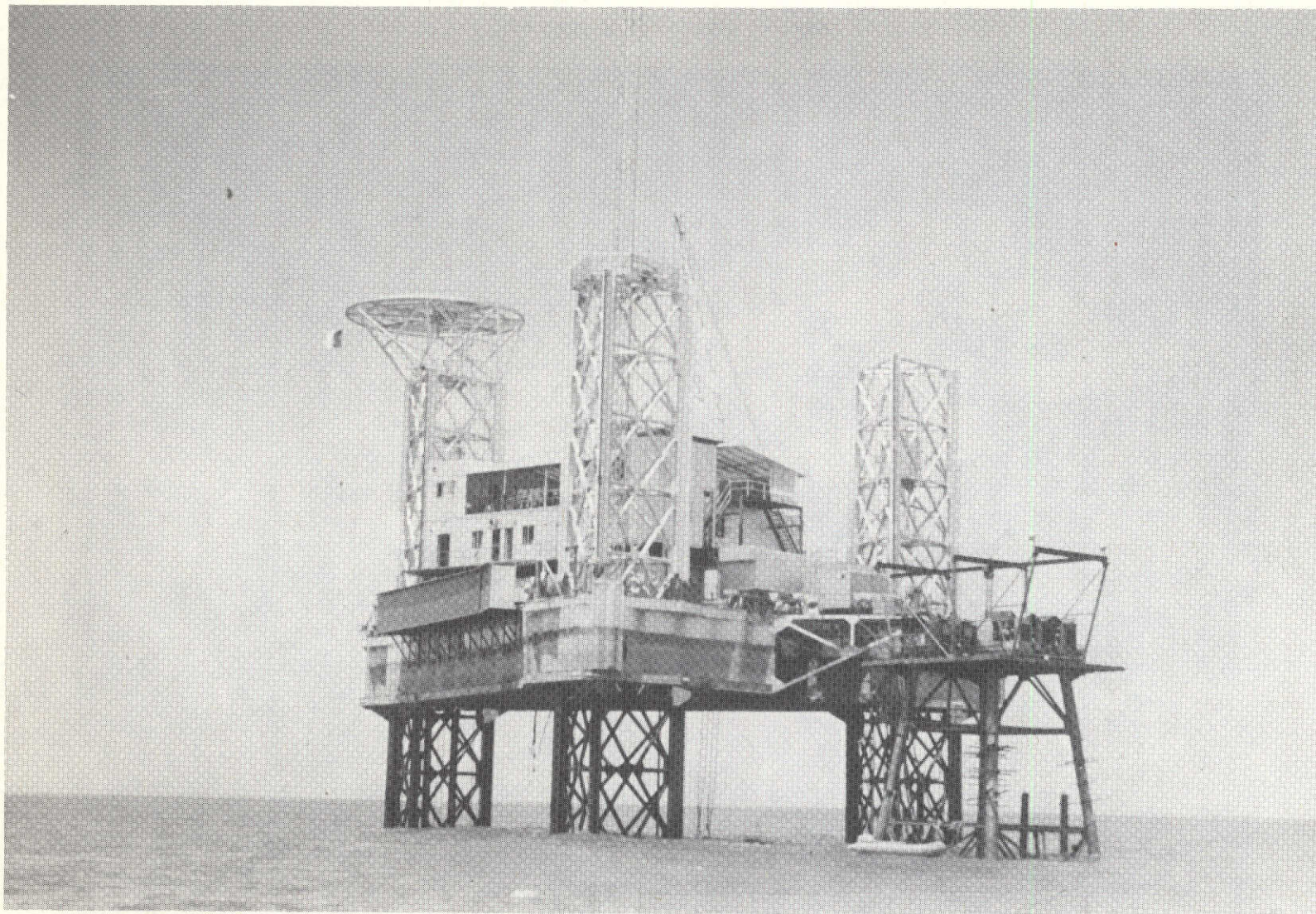


Figure 22. Santa Rita Platform

referred to as "Texas towers." Trailers on the platform house the range control, blockhouse, vehicle, and spacecraft telemetry gear. Also on the platform are the communications stations and facilities for quartering and feeding about 80 persons. A small tower attached to the Santa Rita supports the power generators that supply all electrical power during launch.

The Santa Rita is located about 1700 feet northwest of the San Marco platform. A number of cables for power, equipment control, and communications connect the two platforms.

5.3.2 San Marco Platform

The San Marco launch platform (Figure 23) is a rectangular floatable steel barge, 90 feet wide, 300 feet long, and 13 feet deep, and is normally used by the U. S. Army to establish quick-docking facilities. The barge was leased to Italy and was towed to La Spezia, Italy, in June 1965, for modifications.

The platform was positioned in Formosa Bay at a point 40° 12' 45" east longitude, 2° 56' 18" south latitude, with its length along the east-west direction. A Scout launcher with associated GSE was installed on the west end. The system included:

- The Mark II Scout launcher and shelter
- The vehicle transporter (Figure 24)
- Mechanical and electrical GSE for servicing and testing the vehicle

The San Marco launch-vehicle preparation differed from the WI vehicle preparation in that the San Marco B was assembled on the vehicle transporter under the movable shelter rather than in an assembly building. The same type of assembly-building booms used at WI were installed in the San Marco shelter to permit vehicle checkout on the pad. Two vans, near the shelter, housed the electrical GSE checkout equipment. Electrical cables connected the vans to the shelter.

5.3.3 Communications

A San Marco Project Office, established at Mombasa, was connected to the Santa Rita platform by a CRA radio-voice link (40 watts, 27 to 62 MHz) through Kwale, a repeater station; to the Mobile Italian Telemetry Station (MITS) in Nairobi by landline teletype; and to the NASA STADAN station in Tananarive by radio teletype through Nairobi. Atmospheric conditions made intelligible transmission unpredictable, causing communications to be highly restricted. Walkie-talkie sets were used for communications between the platforms and the base camp.

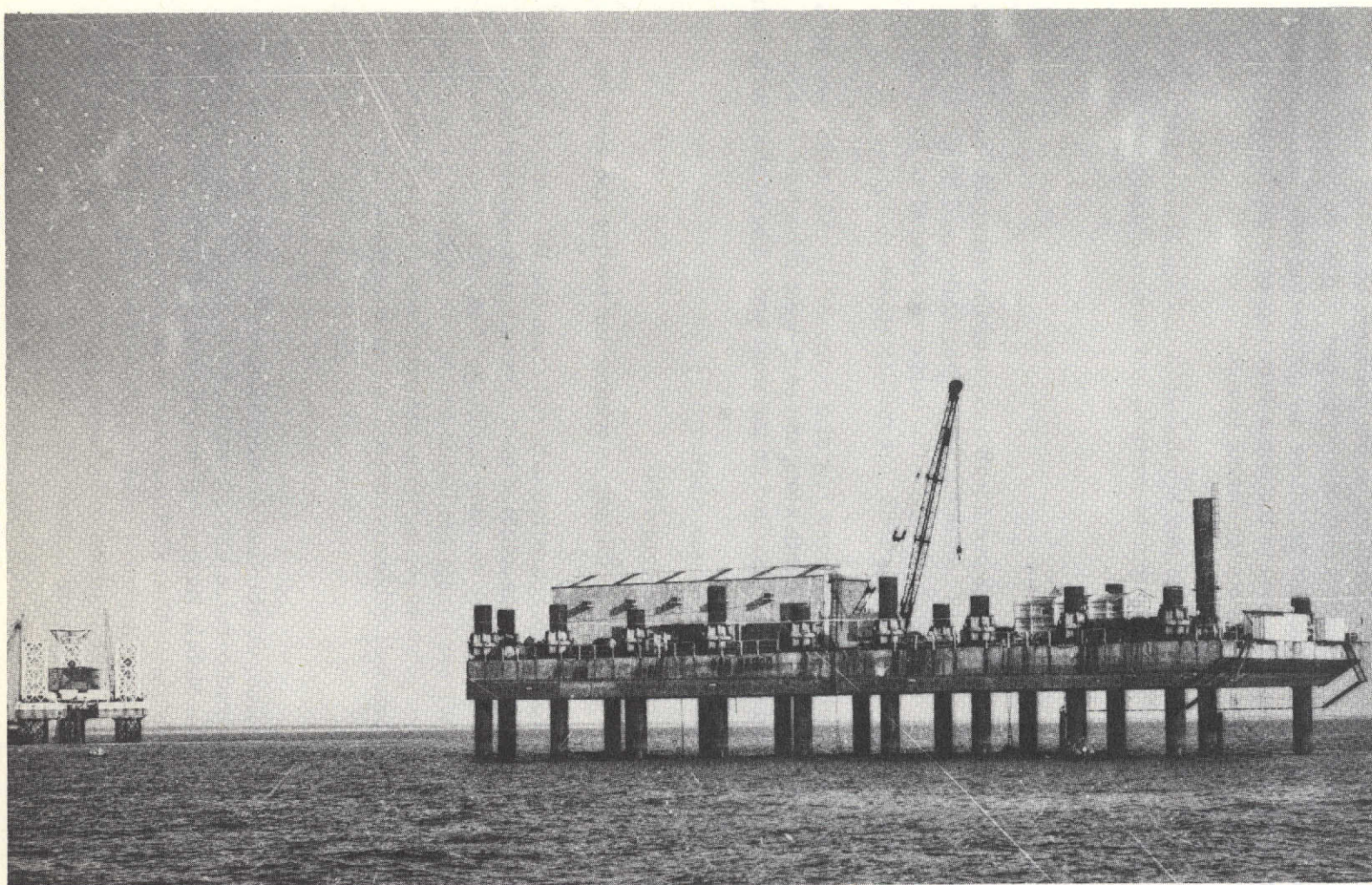


Figure 23. San Marco Platform

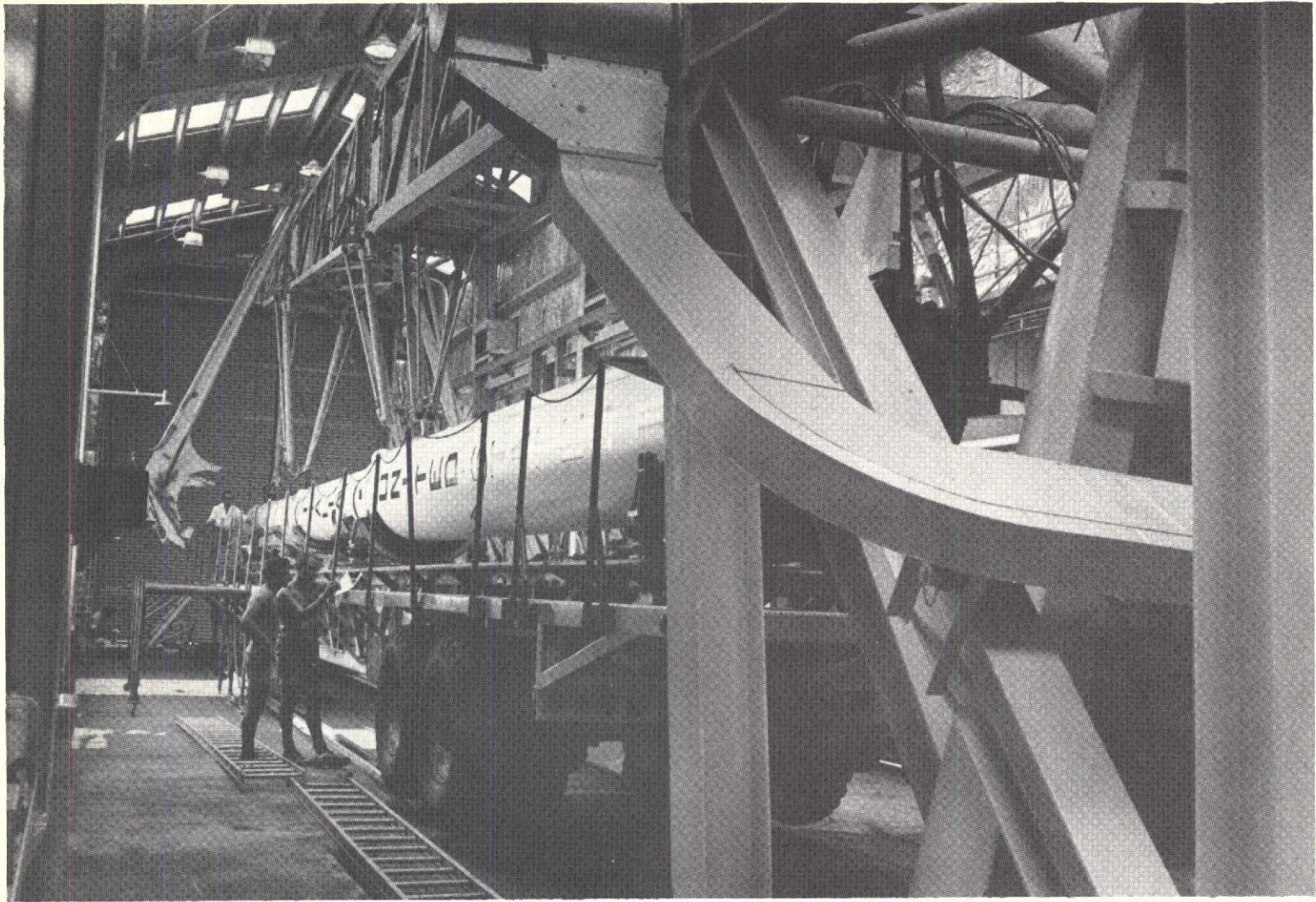


Figure 24. Assembling Scout Vehicle 153-C on-Board Transporter

5.3.4 Down-Range Tracking

The tracking system consisted of the vehicle guidance system and its monitor (the Whittaker gyro system), and one MPS-19 S-band radar (Figure 25) which tracked the vehicle through the first stage and part of the second stage. Guidance and Whittaker gyro data were telemetered through third-stage burn of Scout vehicle S-153C.

5.4 LAUNCH PREPARATION

5.4.1 Dynamic Balancing

Dynamic balancing of the spacecraft/fourth-stage combination posed a problem. In all other Scout launch operations, the fourth-stage/spacecraft assembly was dynamically balanced at the launch site and then incorporated in the vehicle assembly. For the San Marco B, however, the fourth-stage and spin facilities were in the United States, the spacecraft was in Italy, and the launch site was in Africa. CRA, the NASA Centers, and LTV considered a number of iterative techniques regarding the spin-balance operation. To prevent subjecting the flight spacecraft to the rigors of an additional 8000 miles of transportation, and to avoid the inherent difficulty of shipping the fourth-stage/spacecraft as an assembled unit, a decision was made to balance the spacecraft and the fourth stage separately, ship them to the launch site, and then assemble them. Reference 20 contains one of the basic premises on which this decision was made.

5.4.2 Range Operations

A complete set of inert Scout motors was shipped to the platform to permit CRA personnel to assemble a dummy vehicle for training and checkout purposes. After LTV checked out the vehicle transition sections in Dallas (CRA personnel observed the checkout), the sections were shipped by air to Kenya. NASA and CRA personnel checked out the live motors at WI, then shipped them by sea to the launch site. The prototype and flight spacecraft were shipped from Rome through Nairobi to Malindi by air, and then to the platforms by sea. By mid-April 1967, the flight vehicle and spacecraft had been assembled and checked out. A productive launch simulation signalled the completion of the launch preparations.

5.5 PRELAUNCH OPERATIONS

STADAN was assigned the task of tracking the San Marco B spacecraft.²¹ Quito was the primary station selected to implement this task; Lima was scheduled to track each time the spacecraft became visible. Because this constituted 100 percent of the tracking coverage and GSFC had the task of

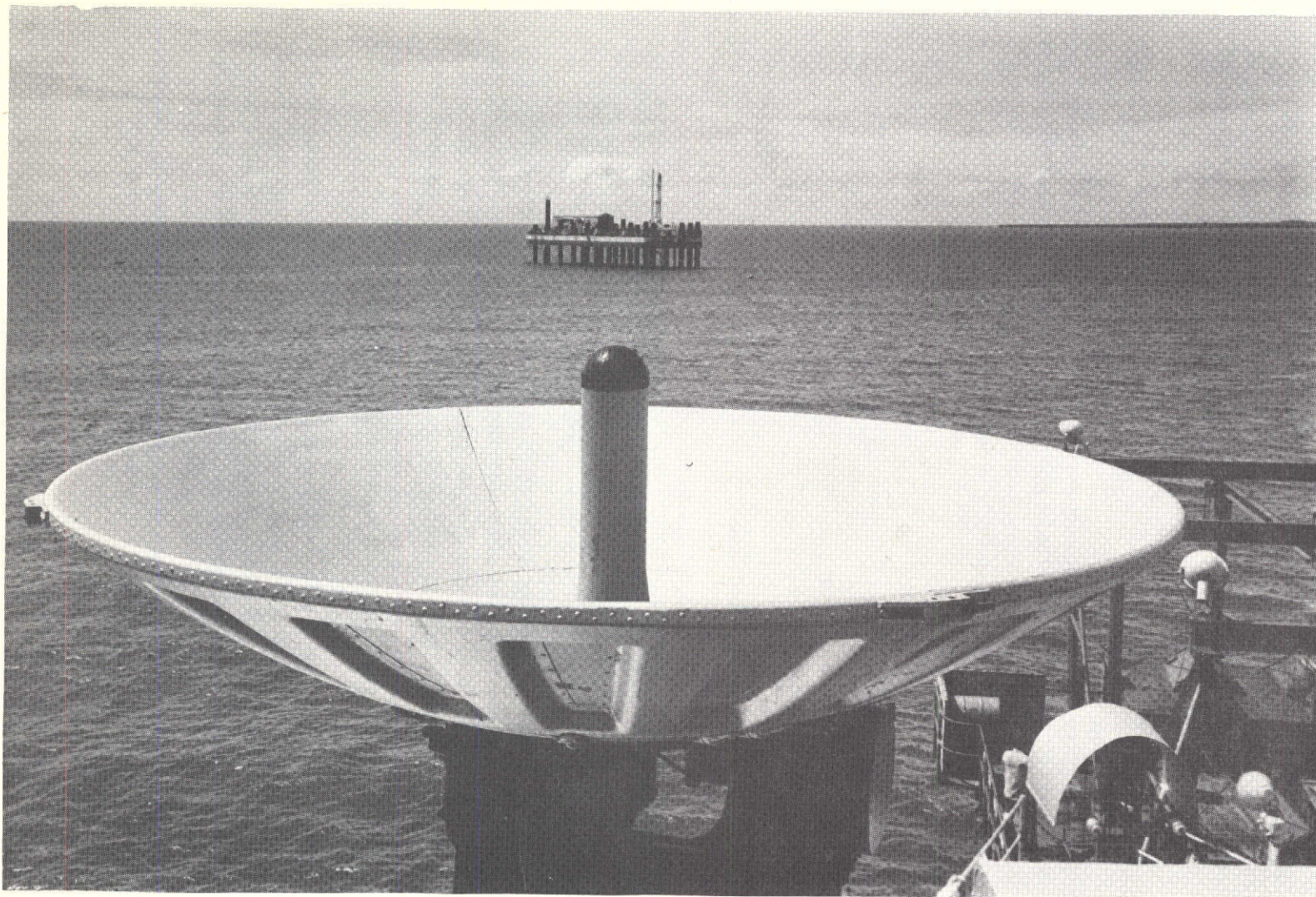


Figure 25. MPS-19 Radar Antenna

determining the orbit, it was decided that substantial communications procedures should be initiated between the Operations Control Center (OPSCON) and Kenya. Accordingly, STADAN was exercised a number of times for launch-simulation procedures. The results showed that the communications system between GSFC and the Kenya launch complex was not suitable or reliable.

5.6 LAUNCH

The San Marco B spacecraft was successfully launched (Figure 26) into an equatorial orbit on April 26, 1967, at 1106 Z by CRA personnel with LRC, WI, GSFC, and LTV on-site personnel acting in an advisory capacity. Orbital parameters for the San Marco 2 satellite (Figure 27) were:

Perigee	218.46 km
Apogee	748.91 km
Period	94.282 min
Semimajor axis	6862.07 km
Eccentricity	0.03865
Inclination	2.89 degrees
Argument of perigee	296.149 degrees
Spin rate	7.5 rpm

Figure 28 shows a typical San Marco 2 orbit.

5.7 TELEMETRY AND TRACKING DATA ACQUISITION

Quito was requested to begin data acquisition after the MITS station at Nairobi centered the three force channels of the air-density experiment. According to the procedures established for data acquisition, apogee passes were defined as passes with a mean height greater than 400 km, and perigee passes less than 340 km. Quito acquired data every other day on all passes having a perigee at one station and a corresponding apogee at the other station.

In scheduling the passes for Quito, GSFC specified, to the MITS station, those passes which were valid according to mutually established procedures. Each station averaged about five telemetry passes per day for a total of 6367 minutes of data. Quito recorded approximately the same number of tracking passes per day. The telemetry-transmitting mode required 10 watts of satellite power; the tracking mode (unmodulated carrier) required only 2 watts of power. On June 26, the ionospheric experiment was turned on and remained on until July 15.

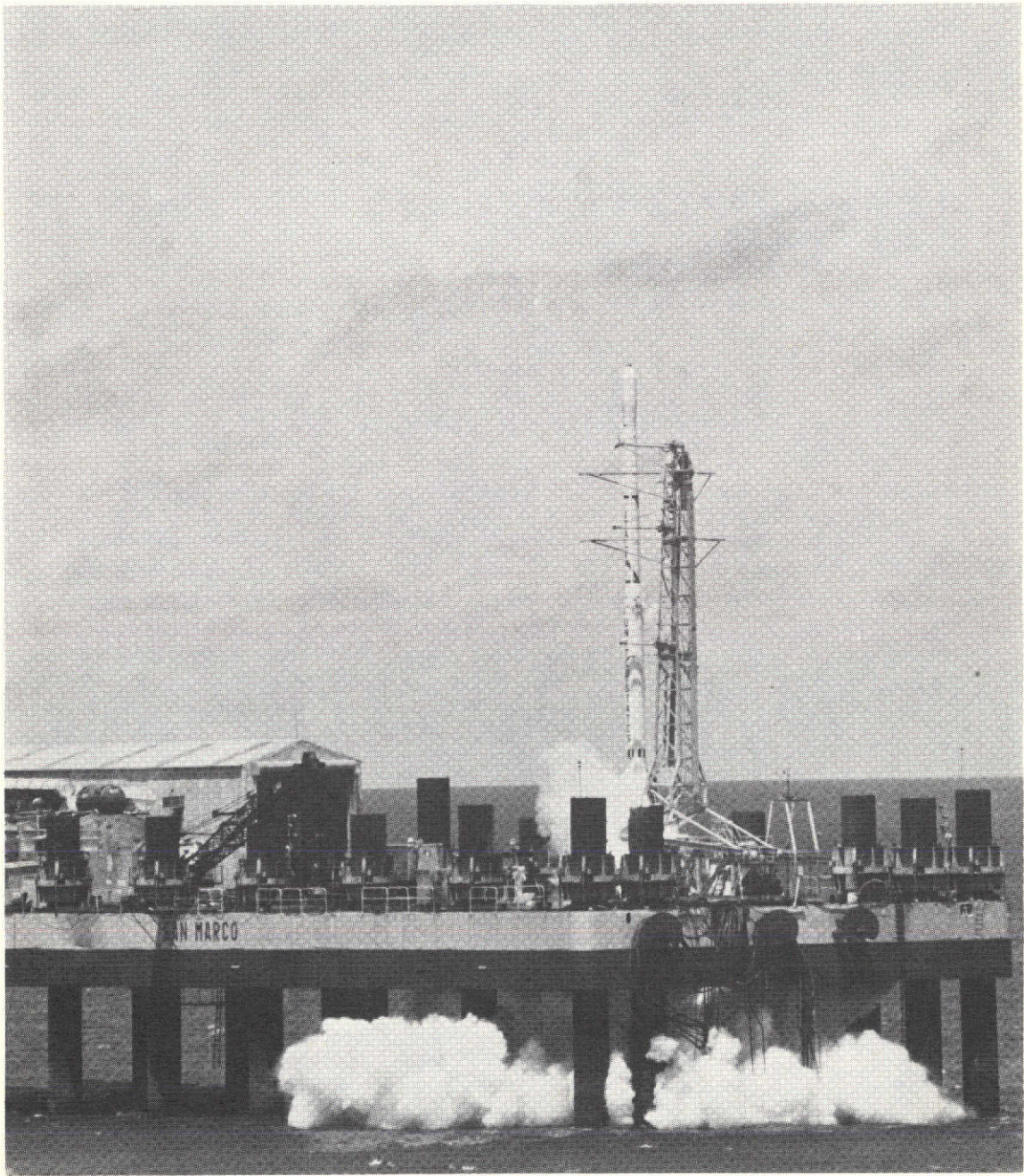


Figure 26. San Marco B Liftoff



Figure 27. San Marco 2 in Orbit

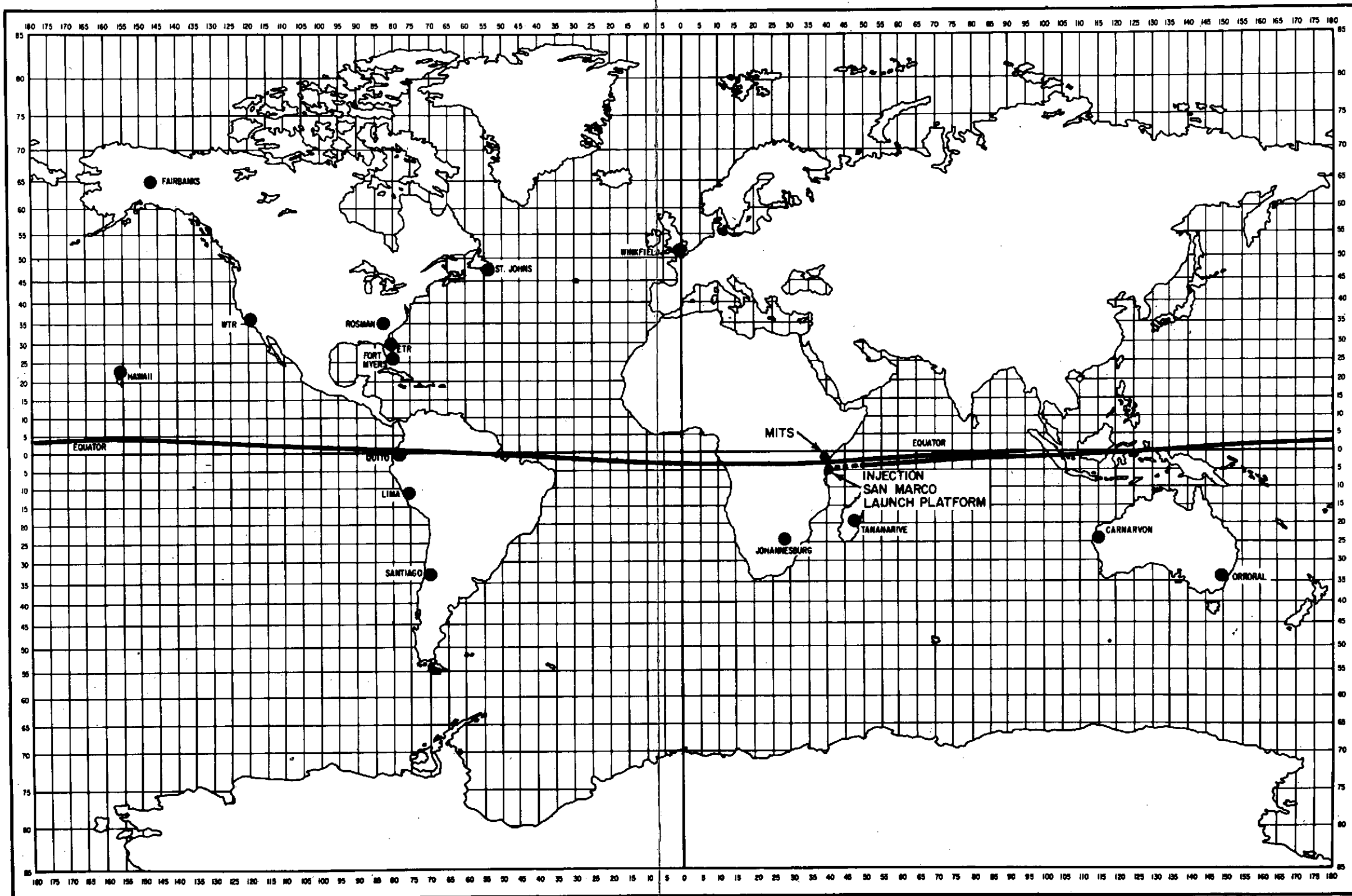


Figure 28. Typical San Marco 2 Orbit

FOLDOUT FRAME

FOLDOUT FRAME

5.8 SPACECRAFT PERFORMANCE

All systems performed normally from launch until August 5, 1967, when the housekeeping telemetry indicated a decrease in the voltage of a 16-volt sector of the power supply. Because this circuit included the command receiver, the continued drop in voltage finally made it impossible to command the satellite. At the last interrogation, August 14, 1967, the spacecraft orbital parameters²² were:

Perigee	207.5 km
Apogee	554.5 km
Period	92.07 min

5.9 REENTRY

The last parameters computed before reentry were:

Perigee	152 km
Apogee	220 km
Period	88.2 min

The estimated impact point of the spacecraft was 0.94 degree north and 343.17 degrees east. Reentry occurred on October 14, 1967, during orbit 2680. The satellite had been in orbit for 171 days. Reference 23 gives further details of the final few days of the San Marco 2 lifetime.

5.10 MISSION RESULTS

5.10.1 Scientific

Although all data had not been processed, Prof. L. Broglio presented representative samples²⁴ to the IX COSPAR International Space Symposium in Tokyo, Japan, in May 1968.

5.10.2 Operational

The successful launch of the San Marco B spacecraft, in April 1967, established the Kenya range as a qualified launch facility. As a result, discussions were initiated on the possible use of the range for future NASA missions.

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6. PROGRAM RESULTS

The objectives of the first San Marco program were fulfilled by the successful investigation of the atmosphere and by the initiation of a launch capability in the equatorial region. As a consequence, the Kenya launch facility is being considered for use by several United States space projects and for San Marco C, another cooperative program.

7. REFERENCES

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APPENDIX A

SEPTEMBER 5, 1962 AGREEMENT BETWEEN

PICCIONI AND JOHNSON

APPENDIX A

ITALY

Outer Space Cooperation: Space Science Research Program

Agreement effected by exchange of notes
Signed at Rome September 5, 1962;
Entered into force September 5, 1962.

The Vice President of the United States of America to the Italian Minister for Foreign Affairs

EMBASSY OF THE
UNITED STATES OF AMERICA
Rome, September 5, 1962

No. 236

EXCELLENCY:

I have the honor to refer to previous conversations between representatives of the United States National Aeronautics and Space Administration and the Italian Space Commission of the National Council of Research regarding cooperation in a scientific experiment which proposes the placement in orbit around the earth of an Italian satellite from an Italian launching facility by means of a rocket provided by the National Aeronautics and Space Administration. The objective of the experiment is to perform measurements of atmospheric and ionospheric characteristics of the earth's atmosphere and to make the resulting scientific data freely available to the world scientific community.

The United States Government confirms the Memorandum of Understanding signed May 31, 1962 by the United States National Aeronautics and Space Administration and the Italian Space Commission, a copy of which Memorandum is enclosed. It is understood that implementation and direction of United States participation in the proposed scientific experiment shall be the responsibility of the United States National Aeronautics and Space Administration and that implementation and direction of Italian participation shall be the responsibility of the Italian Space Commission. The fulfillment and pace of progress of the scientific experiment shall be mutually determined by the two cooperating technical agencies and subject to the conditions which the two agencies have incorporated in the Memorandum of Understanding.

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I have the honor to propose that this Note, together with Your Excellency's reply concurring therein and confirming the enclosed Memorandum of Understanding, shall constitute an Agreement between our two Governments, which shall enter into force on the date of Your Excellency's reply.

Accept, Excellency, the assurances of my highest consideration.

LYNDON B. JOHNSON
Vice-President
of the United States of America

Enclosure:

Memorandum of Understanding
May 31, 1962

His Excellency
ATTILIO PICCIONI,
Minister for Foreign Affairs,
Rome.

APPENDIX B

MAY 31, 1962 MEMORANDUM OF UNDERSTANDING

BROGLIO—DRYDEN

APPENDIX B

MEMORANDUM OF UNDERSTANDING BETWEEN THE ITALIAN SPACE COMMISSION OF THE NATIONAL COUNCIL OF RESEARCH AND THE UNITED STATES NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

1. The Italian Space Commission of the National Council of Research (The Commission) and the United States National Aeronautics and Space Administration (NASA) affirm a mutual desire to conduct a series of experiments which it is hoped will culminate in the launching of a scientific satellite into an equatorial orbit. The objective is to perform measurements of atmospheric and ionospheric characteristics in a region of the earth's atmosphere not previously explored and to make the resulting scientific data freely available. This experimental program is planned to consist of three phases:

(a) First phase—An appropriate sounding rocket will be utilized to provide a flight test of the principal elements of the scientific payload. This launching will take place from the Wallops Island Station and/or from an Italian platform of the San Marco type located near the equator.

(b) Second phase—A prototype of the ultimate satellite payload will be placed in orbit by means of a Scout booster launched from the Wallops Island Station.

(c) Third phase—A scientific satellite, bearing experiments as described above, will be placed in an equatorial orbit by means of a Scout booster launched from a platform of the San Marco type, located in equatorial waters.

2. The cooperating agencies shall proceed from each phase to the next upon mutual agreement that technical feasibility has been demonstrated and, in particular, that environmental requirements for the third phase of the program have been satisfied.

3. The Commission shall, in general, assume responsibility for the following:

(a) Support of Italian personnel for any training required in launching, tracking, data reduction and analysis, and other elements of the program as mutually agreed.

(b) Design, fabrication, and testing of all payloads, including satellite engineering.

(c) Such studies and action as are required to assure a mutually acceptable environment for transport, handling, and launching of the Scout in the third phase of the program.

- (d) The availability, equipping, maintenance, and operation of the "San Marco" towable platforms.
- (e) The establishment of a suitable launch complex for the third phase of the program, including range safety provisions, as mutually agreed.
- (f) Launching of the satellite in the third phase of the program.
- (g) Data analysis in all phases of the program.
- (h) Tracking and data acquisition facilities required in Phase III that are particular to Project San Marco and which are not available from NASA.
- (i) Support, logistics, and all other costs peculiar to Project San Marco.

4. The NASA shall be responsible, in general, for the following:

- (a) Provision of an appropriate sounding rocket and backup, as mutually agreed, for the first phase of the program.
- (b) The provision of Scout boosters with backups for the second and third phases of the program.
- (c) Such training of Italian personnel as may be feasible, and as may be accommodated without significant incremental expense.
- (d) Technical consultation, as appropriate.
- (e) Such additional ground testing of the payloads as may be required.
- (f) The provision of data to facilitate effective design, fabrication, and testing of the payloads.
- (g) Tracking and data acquisition in the first and second phases of the program as can be accomplished by existing NASA sounding rocket and unmanned satellite tracking and data acquisition facilities.
- (h) Provision of tracking and data acquisition services of the Quito, Ecuador, Minitrack Station in Phase three of the program, and such additional communications support at other locations as may be feasible on a noninterference basis, subject to the concurrence, as appropriate, of any foreign governments involved. Special equipment or personnel needed in this connection will be the responsibility of The Commission.

5. No exchange of funds is contemplated between the two cooperating agencies.

6. Each agency agrees to designate a single project manager who shall be responsible for coordinating the agreed functions and responsibilities of each agency with the other. Together they will establish a joint working group with appropriate membership. Details for implementation shall be resolved on a mutual basis within this working group.

7. The scheduling of each of the three phases of the program shall be as mutually agreed.

8. All launches which are a part of this program will be in such areas as may be agreed between the two agencies which shall consult their governments, as appropriate.

9. This Memorandum of Understanding shall be subject to the concurrence of the Italian Foreign Office and the U. S. Department of State, expressed through an exchange of notes.

FOR THE COMMISSION:

/s/ Professor LUGI BROGLIO

GENEVA—May 31, 1962

FOR NASA:

/s/ Dr. H. L. DRYDEN